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Final Report

COMPARATIVE ANALYSIS OF SOLAR THERMIONIC
AND PHOTOVOLTAIC SYSTEMS

Prepared for
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California Institute of Technology
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Pasadena, California
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EOS Report 6937-Final 21 October 1965

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ABSTRACT

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This report presents a summary of the results of a comparative analysis of solar-thermionic and solar photovoltaic power systems performed under JPL Contract 9[illegible]1066 over the period April to September 1965.

Power levels from [illegible] watts to 4 KW are examined in a variety of missions including solar and planetary probes, planetary orbiters and lunar stations. The power source, power conditioning, and battery storage subsystems are considered. Systems are compared on the basis of weight, area and reliability.

This report summarizes the analysis performed during the contract. Detailed backup information is contained in a number of appendices.

Author

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Summary

An investigation and comparison of solar-thermionic and photovoltaic power systems for several future missions was accomplished. Parameters for comparison included weight, area, and reliability. In addition, special areas of interest were examined including packaging limitations imposed by the Atlas-Centaur and Saturn IB-Centaur launch vehicles, structural design limitations, power conditioning circuitry, test requirements, and many other related items. Reliability analysis was conducted to the parts count level. Weight and area projections were based on estimates of component performance improvement in the 1969-75 time period.

The missions examined were:

1) Solar Probe - to 0.2 AU
2) Earth Orbiter - 500 and 10,000 km
3) Mars Orbiter - 10 hr and 50 hr
4) Venus Orbiter - 10 hr and 50 hr
5) Lunar Orbiter - 10 hr and 50 hr
6) Lunar Station - (daylight operation only)
7) Mars Probe
8) Venus Probe

For all missions examined, it was found that by 1975 the surface area of the concentrator in the solar-thermionic system was less than that of the photovoltaic array (see Table 2-I). Advantages of less surface area include a lower value of disturbing torque due to solar radiation pressure and perhaps better ability to package the system and less difficulty with view factors for other items of spacecraft equipment. A solar concentrator, generator, and supporting structure combination is generally more difficult to package than a flat photovoltaic array; each case must be examined in detail to determine view factor and packaging problems.

The reliability of either system could be brought to acceptable levels (> 0.9) with incorporation of suitable redundancy. Reliability

is defined as providing constant power to the load for one year without failure. In one example, a 1 kW system in 1975, photovoltaic system reliability was raised from 0.88 with no redundant elements to 0.98 by incorporation of additional photovoltaic array, redundant batteries, and redundant power conditioning modules. For the same example, solar-thermionic system reliability was raised from 0.35 to 0.93 by incorporating extra thermionic diodes, redundant batteries, and redundant power conditioning modules. The principal item affecting reliability of the solar-thermionic system was the demonstration of thermionic diode reliability and the resultant reliability of the generator. Sufficient test time and test data must be accumulated to demonstrate diode reliability. In line with the conservative mode of this analysis, a reliability limit of 0.9 in 1969 and 0.95 in 1975 for one year's operation for individual thermionic diodes was used in system analysis.

System weight comparisons show that the solar-thermionic system and photovoltaic system were competitive. The solar-thermionic power source (concentrator, generator, and accompanying structure), will be significantly lighter in weight than the photovoltaic array by 1975 for many missions (see Table I). The power conditioning for the solar-thermionic system will be generally considerably heavier than the power conditioning for the photovoltaic system, primarily due to the need for low voltage dc/dc converters to implement generator redundancy.

As a result, overall system weights were close for most cases. New weight-reducing techniques in solar-thermionic power conditioning, or photovoltaic array structures could significantly change the comparative picture. Demonstration of high thermionic diode reliability would also result in significant solar-thermionic system weight reduction.

The inclusion of attitude control weight does not change the conclusions regarding weight comparison significantly except in the case of solar and Venus probes. In these cases, solar radiation pressure is a dominant perturbing torque and the use of solar-thermionics, with less surface area, results in attitude control weight savings.

TABLE I

TYPICAL 1975 SYSTEM SPECIFIC WEIGHT AND AREA

Conditioned Power to Load Watts / System Type (S-T → Solar-Therm., P → Photovoltaic)	Specific Weight, *lb/kw: 200 W S-T	200 W P	1 KW S-T	1 KW P	4 KW S-T	4 KW P	Approximate Specific Area *W/ft^2 (200 W to 4 KW) System rel. > 0.9: S-T	P
0.2 AU Solar Probe	314	400	207	286	149	265	10.5	4.6
Venus 10 hr Orbiter	611	511	499	401	379	373	6.7	6.3
Venus 50 hr Orbiter	642	540	515	428	452	403	9.1	8.3
Venus Probe	314	325	207	205	149	180	10.5	8.1
Earth-500 km Orbiter	796	671	643	557	492	539	4.7	3.6
Earth-10,000 km Orbiter	508	506	391	388	298	373	8.0	5.4
Lunar 10 hr Orbiter	611	547	489	440	379	414	6.7	5.1
Lunar 50 hr Orbiter	642	[illegible]	516	467	430	445	9.1	7.5
Lunar Station	314	394	[illegible]	230	149	259	10.5	4.8
Mars 10 hr Orbiter	779	737	639	637	514	637	2.5	2.4
Mars 50 hr Orbiter	756	716	617	606	528	597	3.5	3.2
Mars Probe	412	454	294	342	232	327	4.0	3.6

* lb/kw and w/ft^2 refers to conditioned power from system at end of one-year mission

A summary of estimated system specific weights and areas is shown in Table I. Lb/kW and W/ft^2 for each system includes the power source, power conditioning, and required battery. The power source for the photovoltaic system is the solar cell array including structure; the power source for the solar-thermionic system includes the concentrator, generator and associated support and deployment structure.

The specific weights and area given in Table I are for 1975 systems with overall reliability greater than 0.9, achieved with the use of redundancy in the source, power conditioning, and battery. Typically, this results in increases in component specific weight: 15 to 35 percent increase in solar-thermionic source, a 10 percent increase in photovoltaic array, a 30 to 70 percent increase in power conditioning, and a 50 percent increase in the battery (for both systems).

The estimates of system weight are based on conservative estimates of possible extensions of existing technology. Thus, weights in 1975 were not based on "trends" but rather on the use of technology now existing in the development or prototype stage. Examples are the 8-mil solar cell, beryllium solar panel structure, aluminum electroformed concentrator, 25 w/cm^2 thermionic diodes, and other components which were assumed to be used in 1975. This conservatism was coupled with several design ground rules which tend to increase power system weight including the need for relatively stiff deployed structures (resonant frequency > 2 cps and ability to withstand 3 g in a deployed condition), the inclusion of 10 percent safety factors in source design, and the limitation of battery depth of discharge to a maximum of 30 percent.

Estimates published in recent literature include numbers as low as 30 to 50 lb/kW for solar power systems; it must be remembered that the spectacular reductions in system weight predicted for future systems usually refer to the source only; to multikilowatt power levels, and assume the application of liberal ground rules regarding structural dynamics, radiation damage, reliability and other features. New developments on the horizon such as very thin radiation-resistant coatings for solar cells, ultralightweight power conditioning, flexible solar arrays, new lightweight batteries, and others may change the conclusions in this text and should be considered in future studies.

As shown in Table I, solar-thermionic systems show weight advantages for the 1975 0.2 AU solar probe, lunar station, and Mars and Venus probe missions. The two systems are generally competitive for 1975 Venus, Lunar, Earth, and Mars orbiters with photovoltaics lighter in weight at the 200 w level and solar-thermionics becoming lighter in weight at the 4-kw level.

It should be noted that in the comparison of "nonredundant" systems, wherein no effort to improve system reliability was attempted, solar-thermionic systems showed weight advantges for most missions. This emphasizes the importance of the reliability versus weight tradeoff.

The percent of system specific weight attributable to source, power conditioning, and battery for typical cases is shown in Table II. Also show is the w/ft^2 figure for the source and the amount of power the source has to provide to supply a constant level of conditioned power to the load. Table II illustrates the fact that the solar-thermionic source weight is generally predicted to be significantly less than the photovoltaic source.

6 D

Conditioned Power to Load	Source Specific Weight,(1)				
	200W		1 KW		4
System Type	ST	P	ST	P	ST
0.2 AU Solar Probe	41	107	43	121	42
Venus 10 hr Orbiter	41	57	43	66	42
Venus 50 hr Orbiter	42	57	43	66	42
Venus Probe	41	58	42	68	42
Earth 50 hr Orbiter	43	79	43	88	42
Earth 100 hr Orbiter	42	96	42	103	42
Lunar 10 hr Orbiter	41	72	43	82	42
Lunar 50 hr Orbiter	42	70	43	81	42
Lunar Station	41	103	43	117	42
Mars 10 hr Orbiter	94	165	96	178	91
Mars 50 hr Orbiter	91	162	93	178	91
Mars Probe	90	142	93	157	91

(1) kW refers to raw power output from source at
of mission

(2) kW refers to conditioned power to load

(3) System includes redundancy, system reliabili

(4) Source output for solar-thermionic system is
by minimum solar intensity, i.e., Earth or M
and Solar probe missions, solar flux control
from source constant.

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TABLE II

BREAKDOWN OF 1975 SPECIFIC WEIGHT AND AREAS(3)

b/KW	Power From Source, Watts(6)						Approx. Source Specific Area(4)(1) w/ft^2		Power Conditioning Sp. Wt lb/KW(2)						Approx. Battery Specific Weight lb/KW(2)	
W	200W		1 KW		4 KW				200W		1KW		4 KW			
P	ST	P	ST	P	ST	P	ST	P	ST	P	ST	P	ST	P	ST	P
127	400	310	1750	1540	6800	6160	14.2	6.5	200	205	100	70	45	40	32	30
69	640	418	2800	2090	11000	8360	14.2	13.0	275	210	165	82	60	49	204	181
69	460	308	2000	1540	8000	6160	14.2	13.0	250	205	135	70	50	41	296	256
71	400	310	1750	1540	6800	6160	14.2	11.3	200	205	100	70	45	40	32	30
93	930	560	4000	2600	16000	11220	14.2	10.8	350	230	225	92	76	59	248	220
111	520	352	2300	1760	9000	7040	14.2	10.8	250	205	145	74	53	45	150	132
85	640	418	2800	2090	11000	8360	14.2	10.8	275	210	165	82	60	49	204	187
85	460	308	2000	1540	8000	6160	14.2	10.8	250	205	135	70	50	42	296	272
123	400	310	1750	1540	6800	6160	14.2	10.8	200	205	100	70	45	40	32	30
194	640	418	2800	2090	11000	8360	5.4	5.0	275	210	165	82	60	49	204	182
190	460	308	2000	1540	8000	6160	5.4	5.0	250	205	135	70	50	42	296	262
167	400	310	1750	1540	6800	6160	5.4	5.0	200	205	100	70	45	40	32	30

end

ty > 0.9

determined
ars. Venus
keeps power

(5) All calculations assume that conditioned power to load is constant throughout mission.

(6) Power from source is greater than required conditioned power to load due to power conditioning losses, energy storage losses, and extra redundant sources to increase source reliability. Power from source refers to power on line prior to any power conditioning.

1. PROGRAM OBJECTIVES AND GROUND RULES

The study program consisted of the following basic steps:

1. Definition of the system and mission model including component performance improvements anticipated over the 1967 to 1975 period.
2. An analysis of the performance of each type of power subsystem over the range of 100 watts to 4 kw for solar and planetary probes; earth orbiters; lunar, Mars and Venus orbiters; and lunar stations.
3. A comparison of each type of power system on the basis of weight, area and reliability for the missions and time period above.

The definition of the system model in some areas has relied upon data supplied by JPL for the future performance of components. Other component performance information was derived from previous studies performed by EOS and other organizations.

This section summarizes some of the major criteria which define the scope of the comparative analysis. Detailed criteria necessary for analysis are contained in the Appendices to this report. Major ground rules included:

1. <u>Launch Date</u>

   Launch dates from 1969 through 1977 were investigated. It was assumed that component technology must be available two years prior to launch.

2. <u>Missions</u>

   The missions studied are:

a. Sun probes to ranges within 0.2 AU

b. Planetary flyby probes to Venus and Mars

c. Earth orbiters for circular equatorial orbits of 500 and 10,000 km

d. Lunar, Mars and Venus orbiters for 10 and 50 hr eccentric orbits

e. Lunar station operation during the daylight hours

3. Life

The desired lifetime was assumed to be a one-year period (8700 hrs). For the planetary orbiters, two cases were considered; in the first case part of the year was spent in transit, in the second case the entire year was spent in orbit. Weight numbers in Tables I and II refer to the first case.

4. Power Output

Power levels ranging from 200 watts to 4 kilowatts were considered. It was assumed that the power level should be constant throughout the mission. For example, on a 0.2 AU solar probe, the power at earth would be equal to the power near the sun. For solar probes, this assumption has considerable effect on power subsystem design and should be reviewed to determine whether a variable power output, increasing at close distances to the sun, would enhance vehicle design. The power output was assumed to be 50 percent dc and 50 percent ac with a constant load requirement. The energy storage requirement was assumed dependent on the orbital shadow time or, in the case of probes, on a nominal maneuver requirement of 1-1/2 hrs.

5. Structures

The solar array (photovoltaic or thermionic) was assumed to be rigidly attached to the vehicle for all cases under consideration. Structural technology for the photovoltaic panel was based upon the use of thin gauge sheet metal structures and lightweight truss frame support similar in concept to the Mariner IV solar panels. Structural design was based on reasonable criteria such as a minimum resonant frequency of 2 cps, a 3 "g" load for retromaneuvers for orbiters, and typical vibration loads during launch. The 1975 structural weight was estimated using conservative judgments regarding technological advances. Similar technology was assumed for the solar-thermionic system support.

6. Booster/Shroud/Spacecraft

The two boost vehicles considered were the Atlas-Centaur and Saturn IB-Centaur. Packaging studies were based on the use of projected shrouds for these vehicles and reasonable judgments with regard to the type of spacecraft which would be used, depending on payload limitations and available power.

7. Component Performance

It was assumed that one-piece solar concentrators up to 15 feet in diameter would be available which would be electroformed and made from lightweight nonmagnetic materials such as aluminum, beryllium and magnesium. Aluminum mirrors 9.5 feet in diameter would be available for the 1969 flight date and 15-foot aluminum mirrors would be available for a 1971 flight date. Beryllium and magnesium mirrors up to 15 feet in diameter would be available for a 1973 flight date. Also, cesium reservoir controls would be available for a 1970 flight date and solar flux controls for a 1970 flight. Many assumptions regarding component performance were required to complete the analysis and are discussed in later sections. Significant ground rules are:

a. The maximum mirror diameter available will be 9.5 feet for 1968-1969 flight dates and 15 feet for 1971-1977 flight dates.

b. Concentrator specific weights will decrease with an assumed concentrator specific weight of 0.8 lb/ft^2 in 1969 and 0.5 lb/ft^2 in 1977 at the maximum mirror diameters.

c. Steadily improving generator efficiencies ranging from 10 percent in 1969 (at 1700°C) to 22.4 percent in 1977 (at 1700°C).

d. An improvement in solar cell efficiency; for a 1969 flight date the nominal cell output will be 10.9 percent efficiency (at 28°C) while in 1977 the output will be 11.3 percent. Also, solar cell thickness and weight will decrease with 12-mil-thick cells being used in 1969 and 8-mil thick cells being used in 1977.

Other assumptions regarding component performance are summarized in the text of the report. Many of the component performance parameters are based on judgment regarding improvements which can be obtained by consistent development efforts.

8. System Components

A summary of the basic components of a solar-thermionic system is shown in the block diagram of Fig. 1-1 along with energy losses in the system. A detailed explanation of component performance is contained in Ref. 1*, below.

Figure 1-2 shows several items of prototype hardware of the solar-thermionic system under development, and a typical system configuration.

Figure 1-3 shows the basic block diagram of the photovoltaic system.

Figure 1-4 shows typical photovoltaic system components. The Mariner vehicle illustrates the manner in which the solar panels are attached, and also shows a typical photovoltaic array on a spacecraft.

This report does not consider detailed operation of each component, but rather concentrates on system performance and comparison.

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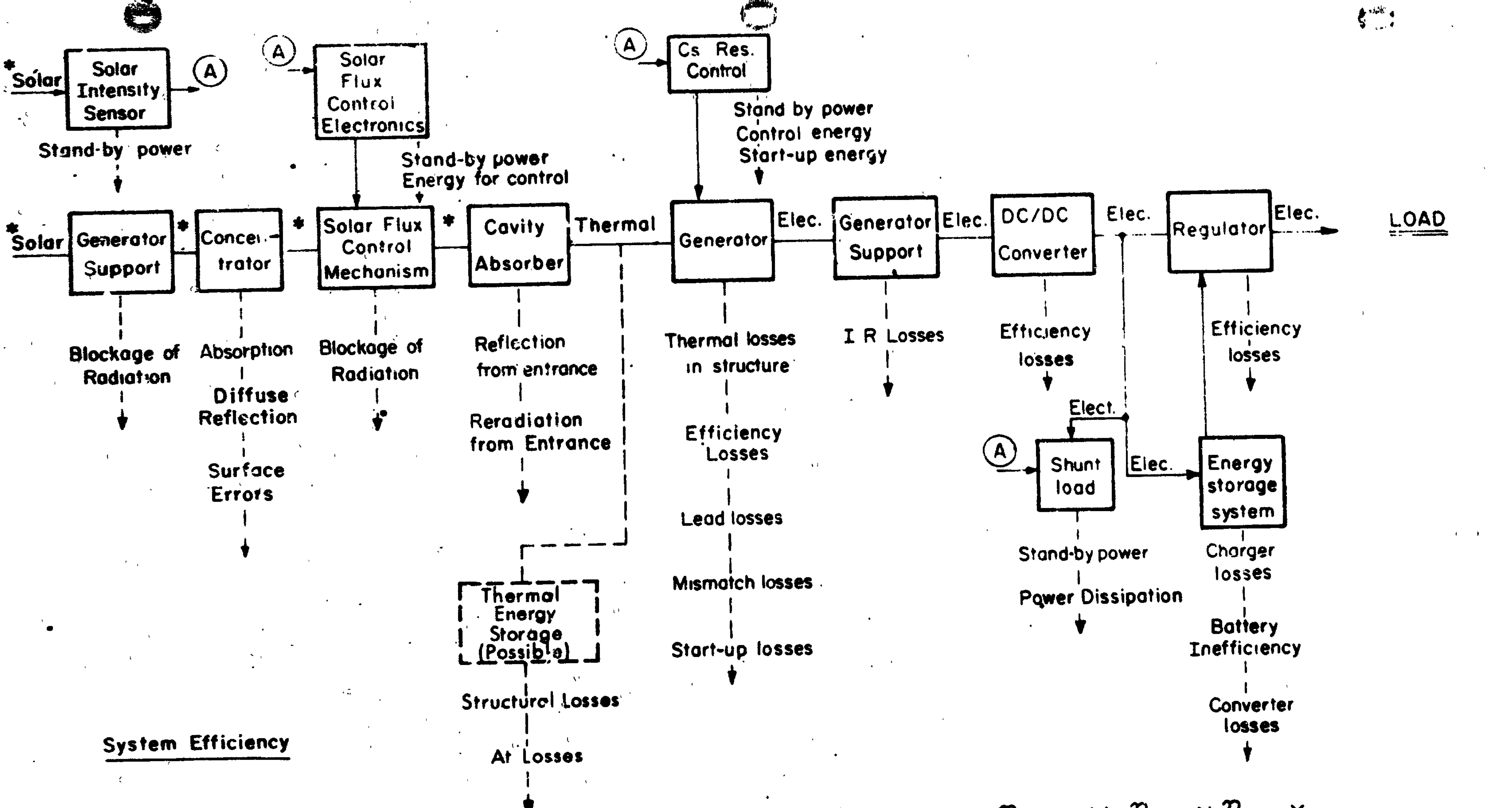
*Final Report on "Analysis of Ancillary Equipment for Solar Thermionic System," 10 March 1965, prepared under JPL Contract 950699.



System Efficiency

$$\eta_{sys} = \underset{\text{Obscuration}}{\underline{\eta_{obscr}}} \times \underset{\text{Concentrator-Absorber Efficiency}}{\underline{\eta_{c-a}}} \times \underset{\text{Generator Efficiency}}{\underline{\eta_{gen}}} \times \underset{\text{Lead eff.}}{\underline{\eta_{lead}}} \times \underset{\text{DC/DC Converter Efficiency}}{\underline{\eta_{conv}}} \times \underset{\text{Regulator Efficiency}}{\underline{\eta_{reg}}} \times \underset{\text{Control Efficiency}}{\underline{\eta_{cont}}} \times \frac{1}{1 + K/\tau_L \eta_{stor}}$$

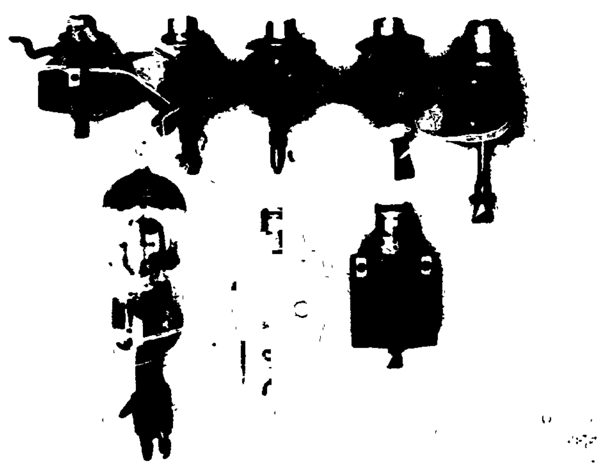
(K = Factor depending on time in dark, peak power, start-up time, etc.
τ_L = Light time η_{stor} = Storage efficiency)

FIG. 1-1 SOLAR-THERMIONIC SYSTEM BLOCK DIAGRAM AND ENERGY LOSSES

PROTOTYPE THERMIONIC CONVERTER WITH 5 THERMIONIC DIODES

TYPICAL SYSTEM CONFIGURATION

9-1/2' CONCENTRATOR MOUNTED FOR TEST



PROTOTYPE THERMIONIC CONVERTERS, 1962-65

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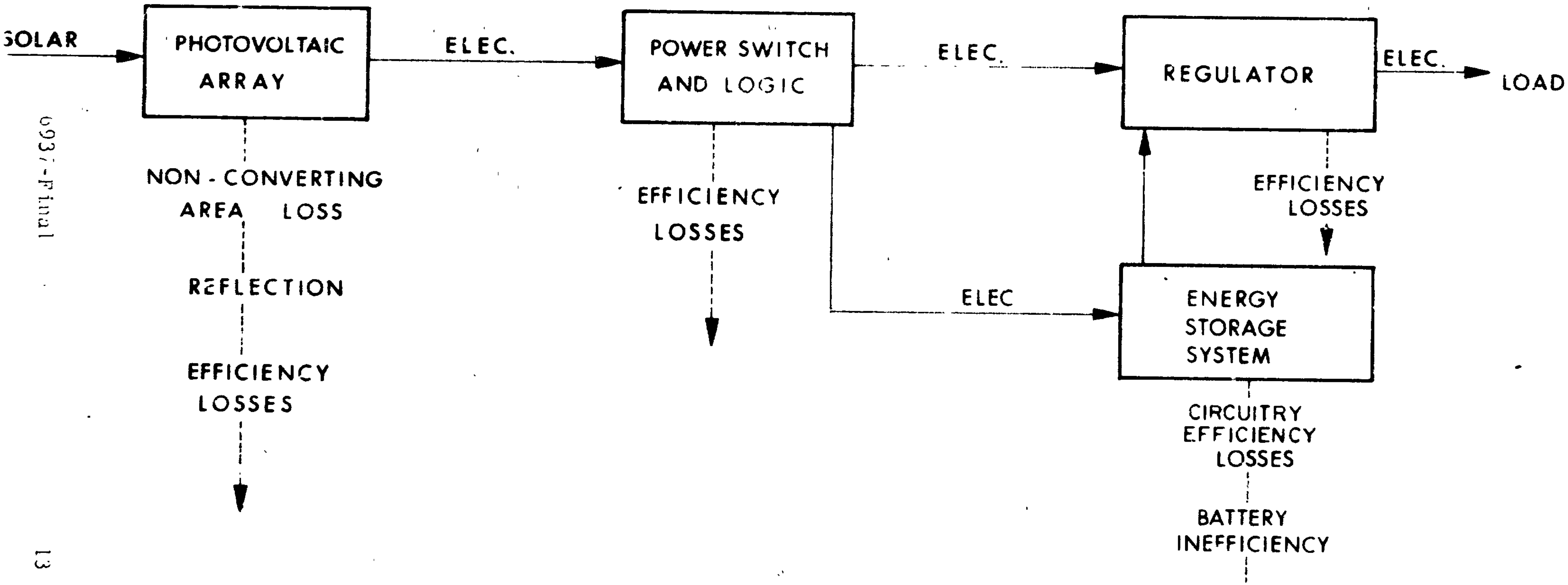
5 ft. CONCENTRATOR, GENERATOR SUPPORTS, AND PROTOTYPE GENERATOR

FIG. 1-2 SOLAR-THERMIONIC SYSTEM PROTOTYPE COMPONENTS



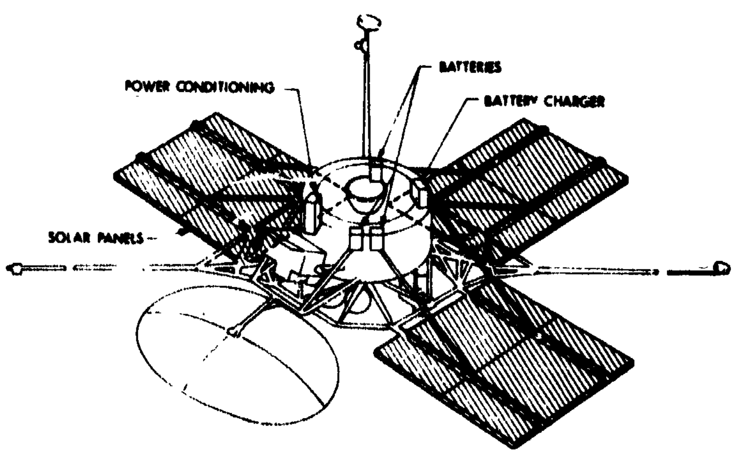
$$\underset{\text{SYSTEM}}{\eta_{sys}} = \underset{\text{ARRAY EFFICIENCY}}{\underline{\eta_{array}}} \times \underset{\text{PS \& L EFFICIENCY}}{\underline{\eta_{PS\&L}}} \times \underset{\text{REGULATOR EFFICIENCY}}{\underline{\eta_{reg}}} \times \frac{1}{1 + K/\tau_L\,\eta_{stor}}$$

(K = FACTOR DEPENDING ON TIME IN DARK, PEAK POWER START-UP TIME, ETC.
τ_L = LIGHT TIME η_{STOR} = STORAGE EFFICIENCY)

FIG. 1-3 PHOTOVOLTAIC SYSTEM BLOCK DIAGRAM AND ENERGY LOSSES

Ranger Power Subsystem Including
Solar Panels, Battery, Electronic
Modules

Mariner C Vehicle Showing
Photovoltaic Panels



Typical Power System Configuration

FIG. 1-4 TYPICAL PHOTOVOLTAIC COMPONENTS

2. SYSTEM COMPARISON

A detailed study of the expected weight, area and reliability of the solar-thermionic and photovoltaic systems was accomplished and is summarized below.

2.1 Area

Table 2-I shows an area comparison for thermionic and photovoltaic systems for the missions of interest in the years 1969 and 1975. Area refers to the projected area of the solar concentrator or photovoltaic array. The numbers presented refer to a non-redundant system, i.e., a system in which no redundant modules are incorporated to improve system reliability.

Figure 2-1 graphically illustrates an area comparison for a 1 KW system for several missions. Two areas are shown; the non-redundant system and a redundant system in which extra array area is included to increase power source reliability. For the photovoltaic system, the array reliability is increased to a level exceeding 0.99 by increasing area about 10 percent. For the thermionic system, solar-thermionic system reliability is assumed to have been increased to about 0.9, based on calculations and circuitry discussed in later paragraphs.

As shown, in 1969 it is expected that the solar-thermionic system would require more area than the photovoltaic system except for the solar probe mission.

However, by 1975 the expected increases in solar-thermionic source efficiency will result in considerable decreases in concentrator area. The solar-thermionic system will require less surface area than the photovoltaic system for all missions considered in 1975. This is true even when considerable redundancy is added to the solar-thermionic array, where typically the solar concentrator projected area is increased by 30 percent.

TABLE 2-

AREA COMPARISON FOR THERMIONIC AND

Thermionic System Area (

Watts[(1)]	Mars-10 hr 1969	Mars-10 hr 1975	Mars-50 hr 1969	Mars-50 hr 1975	Lunar-10 hr 1969	Lunar-10 hr 1975	Lunar-50 hr 1969	Lunar-50 hr 1975	Earth-500 Km 1969	Earth-500 Km 1975	Earth-10,000 1969	197
100	66	31	48	22	25	12	18	8	36	17	20	
200	132	62	96	45	50	23	36	17	72	33	40	1
400	264	124	192	90	100	46	72	34	144	66	80	[illegible]
700	462	217	336	157	175	81	126	59	252	115	140	6
1000	660	310	[illegible]	224	250	115	180	84	360	165	200	9
2000	NA	620	[illegible]	448	500	230	360	168	720	330	400	1
4000	NA	NA	NA	[illegible]	[illegible]	460	720	336	NA	660	800	3

Photovoltaic Syst

Watts[(1)]	Mars-10 hr 1969	Mars-10 hr 1975	Mars-50 hr 1969	Mars-50 hr 1975	Lunar-10 hr 1969	Lunar-10 hr 1975	Lunar-50 hr 1969	Lunar-50 hr 1975	Earth-500 Km 1969	Earth-500 Km 1975	Earth-10,000 1969	197
100	44	38	32	29	22	18	16	13	26	25	18	
200	89	77	65	57	43	36	31	27	52	50	36	
400	177	154	129	114	86	72	62	53	102	100	71	
700	310	269	226	[illegible]	151	126	109	93	179	175	[illegible]	[illegible]
1000	443	384	323	[illegible]	215	180	155	133	256	250	179	
2000	886	769	645	571	430	360	311	266	512	500	357	
4000	1773	1537	1291	1143	861	718	622	532	1023	1005	715	

Notes

(1) Refers to Conditioned Watts to Load

NA indicates that system could not be packaged in Saturn IB/Centaur

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OVOLTAIC SOLAR ARRAY

Lunar Station 1969	Lunar Station 1975	Mars Probe 1969	Mars Probe 1975	Solar Probe 1969	Solar Probe 1975	Venus-10 hr 1969	Venus-10 hr 1975	Venus-50 hr 1969	Venus-50 hr 1975	Venus Probe 1969	Venus Probe 1975
16	7	42	20	16	7	25	12	18	8	16	7
32	15	84	40	32	15	50	23	36	17	32	15
64	30	168	80	64	30	100	46	72	34	64	36
112	51	294	140	112	51	175	81	126	59	112	51
160	73	420	200	160	73	250	115	180	84	160	73
320	146	840	400	320	146	500	230	360	168	320	146
640	292	NA	800	640	292	1000	460	720	336	640	292

)

22	19	29	25	23	20	18	15	13	11	12	11
43	38	58	51	46	39	35	29	26	22	24	22
87	76	115	101	92	78	71	58	52	44	48	45
152	133	202	177	162	137	124	102	90	76	84	79
218	190	289	253	231	195	177	146	129	109	120	112
434	379	577	505	462	391	355	292	258	218	241	225
868	758	1155	1012	924	782	709	583	516	436	482	450

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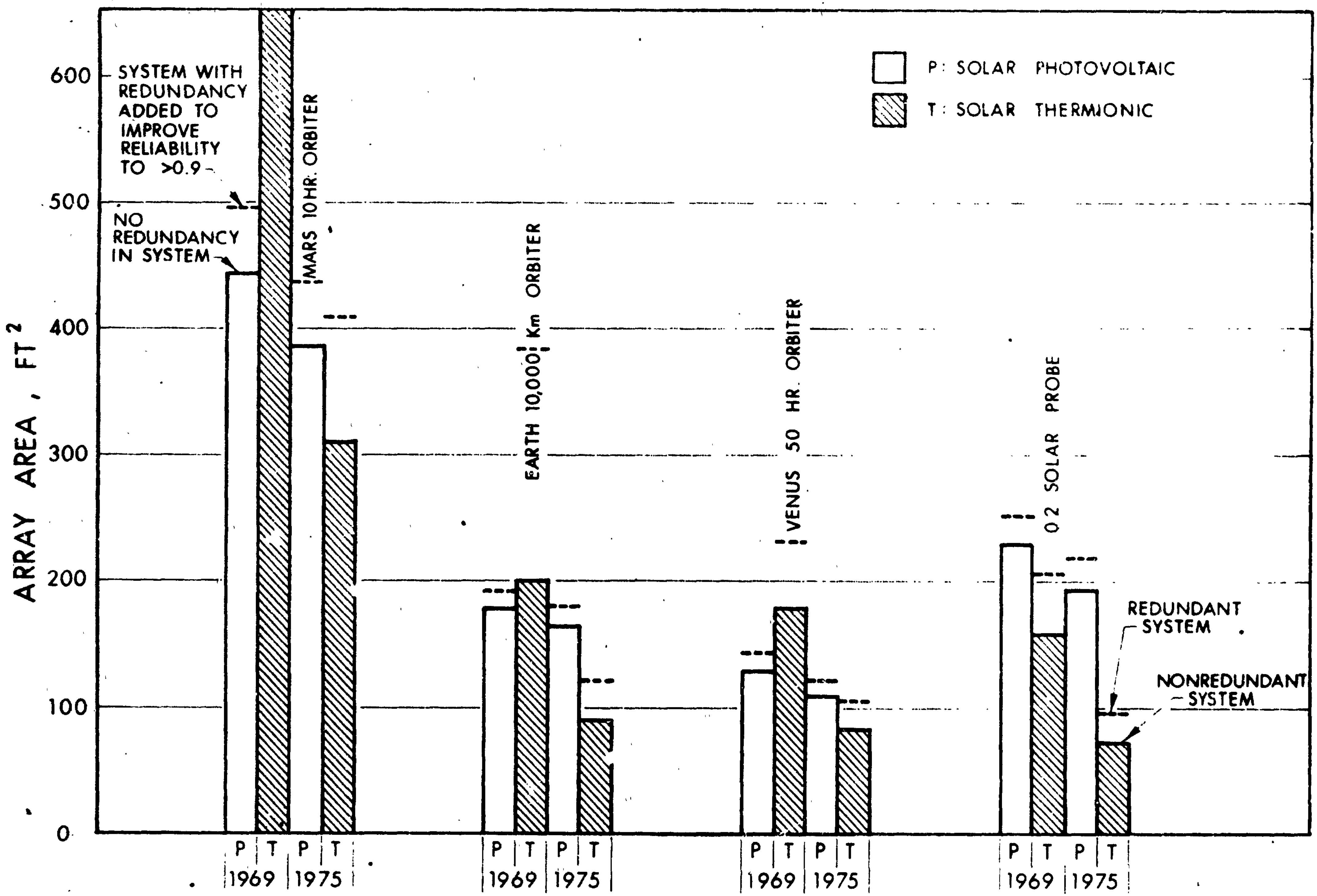


FIG. 2-1 AREA COMPARISON FOR THERMIONIC AND PHOTOVOLTAIC SOLAR [illegible]

The area advantage of the solar-thermionic system is particularly striking in the solar probe mission where for a 1 KW load in 1975, 73 ft^2 is required for thermionics versus 195 ft^2 for the photovoltaic system in the nonredundant case.

The advantage of solar-thermionics with regard to area is due primarily to the high source efficiency, typically 15 to 18 percent in 1975. This compares with a maximum photovoltaic array efficiency of typically 10 to 11 percent.

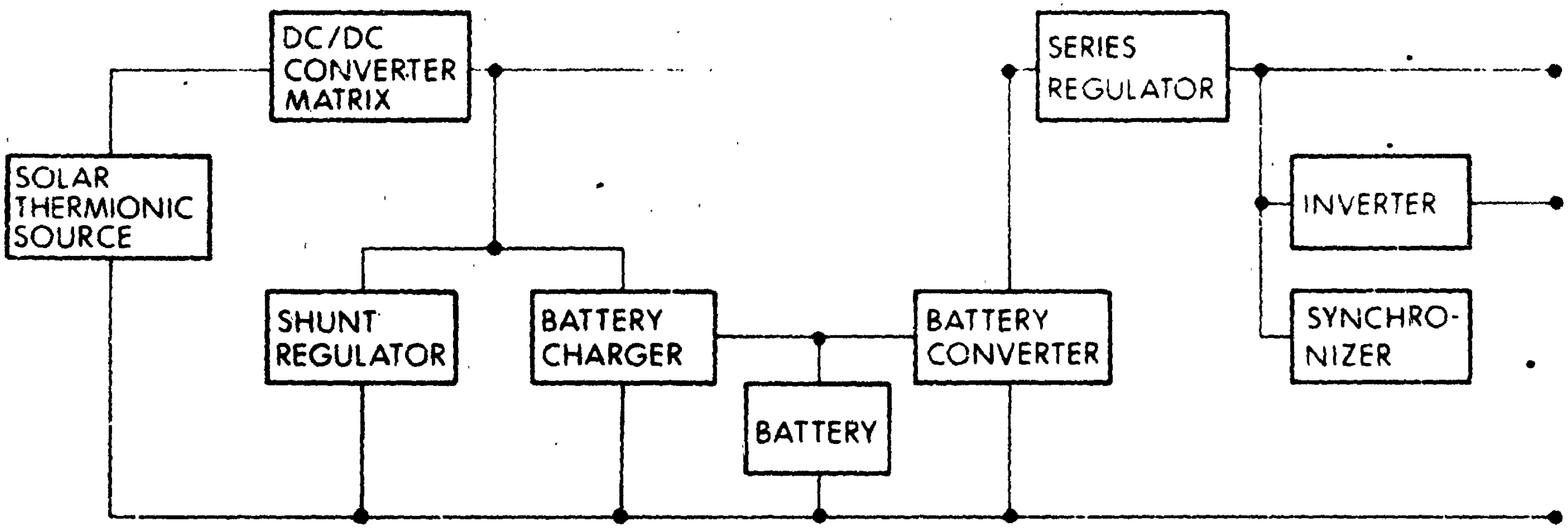
For Venus missions (one case is shown in Fig. 2-1), the watt output per unit area of the solar-thermionic source must be based on earth conditions. Excess solar radiation at Venus or closer distances to the sun must be eliminated via flux control or other means. For photovoltaic systems, the watt output per unit area will vary in accordance with solar intensity and radiation degradation. These factors control the temperature of the array, the selection of cover glass, and (at distances from the sun of roughly less than 0.5 AU) the optimum angle of incidence of the photovoltaic array to the solar radiation. For an 0.2 AU solar probe, the w/ft^2 figure for the photovoltaic array is less than that achievable at earth due to solar proton degradation over the mission duration and the effect of high intensity solar radiation on array I-V curves.

All area calculations are based on a one-year mission, accounting for solar panel degradation due to solar protons (and Van Allen for earth) with an optimum cover glass. Solar-thermionic concentrator reflectivity of 0.85 was assumed throughout the mission with no degradation.

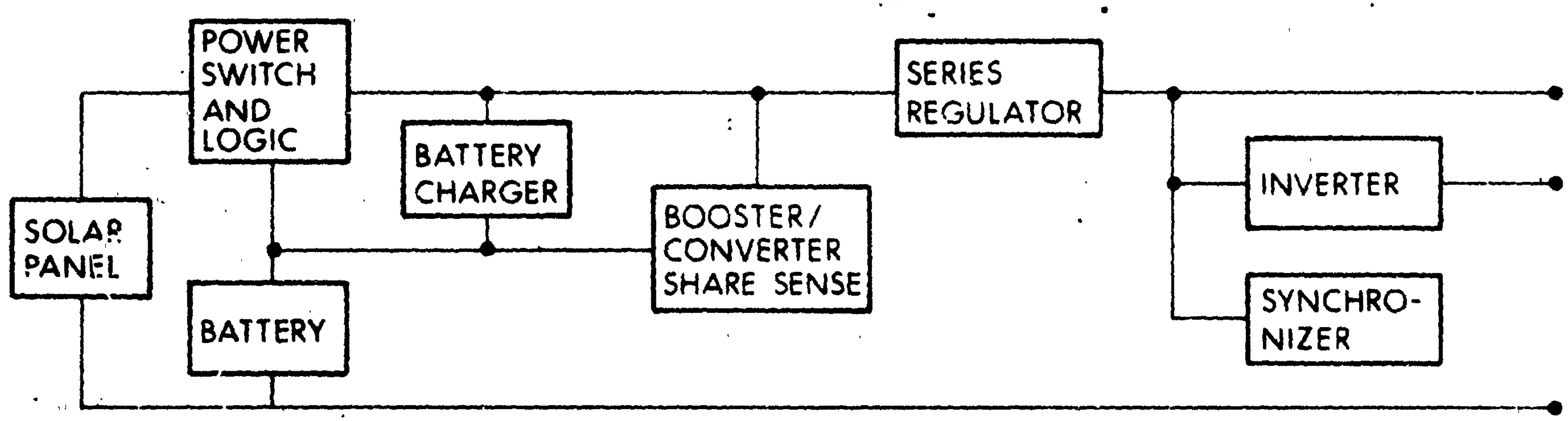
At the 4000-watt load level and 2000-watt level, several cases were found for both 1967 and 1975 flights where it was not thought practical to package the solar-thermionic system within the limits of the Atlas-Agena or Saturn IB/Centaur shroud. These cases are marked by NA in Table 2-I. Although the photovoltaic system needs more area, the flat panel configuration is more amenable to packaging than the concentrator and generator support of the thermionic system.

2.2 Reliability

An analysis of photovoltaic and solar-thermionic system reliability was accomplished using the basic block diagrams of Fig. 2-2.



A. SOLAR-THERMIONIC SYSTEM



B. PHOTOVOLTAIC SYSTEM

FIG. 2-2 BASIC SYSTEM BLOCK DIAGRAMS

Power conditioning reliability calculations were made using detailed parts counts of each electronic module (these varied with power level) and "minuteman" failure rate data. A reliability of 0.98 was assumed for the battery based on sketchy data derived from the Gemini, Ranger, Mariner, and other programs. Detailed calculations were performed to estimate thermionic generator and photovoltaic panel reliability using assumed failure rates for thermionic diodes and individual solar cells; for solar cells, failure rates of 0.1×10^{-6} were used while the assumed reliability of thermionic diodes for one year's operational life was 0.9 in 1969 and 0.95 in 1975. Detailed explanations for these assumptions are contained in the Appendices for this report.

Table 2-II shows a typical reliability summary for parts of the system. Power conditioning reliability is calculated from a "single-thread" type of analysis of all electronic modules. Because of increased stress level and increased number of parts, the reliability of the power conditioning generally decreases with increased power level.

Structural reliability was not considered in detail. However, sufficient design margin was allowed in weight assumptions for relatively high reliability in structures. Structural reliability was not considered in system reliability estimates because it was assumed to be high relative to other system components.

System weight and area calculations were based on two general cases. The nonredundant system used no extra electronic modules or sources (such as thermionic diodes) to increase reliability.

A redundant system has the following features:

1. Redundant power conditioning modules with failure sense and switching units. Redundant modules in the photovoltaic system are the series regulator (2)*, inverter (2), and battery charger (3). Redundant modules in the solar-thermionic system are the series regulator (2), inverter (2), battery charger (3), shunt regulator (3) and dc/dc converters (one converter for every 100 or 200 watts from source).

*refers to number of redundant modules

TABLE 2-II

TYPICAL RELIABILITY SUMMARY

		Non-Redundant	With Redundancy
A. Solar-Thermionic Components			
1. Source - Concentrator/Structure	-	High	High
Generator*	-	$(R_D)^N$	~ 0.9
2. Battery	-	~ 0.98	~ 0.995
3. Power Conditioning	-	~ 0.94 at 200 watts	~ 0.990
		~ 0.89 at 1 KW	~ 0.985
		~ 0.85 at 4 KW	~ 0.98
B. Photovoltaic Components			
1. Source - Structure	-	High	High
Solar Cells	-	~ 0.95	> 0.99
2. Battery	-	~ 0.98	~ 0.995
3. Power Conditioning	-	0.96 at 200 watts	~ 0.994
		0.947 at 1 KW	~ 0.991
		0.93 at 4 KW	~ 0.987

*R_D is reliability of an individual thermionic diode and N is the number of diodes.

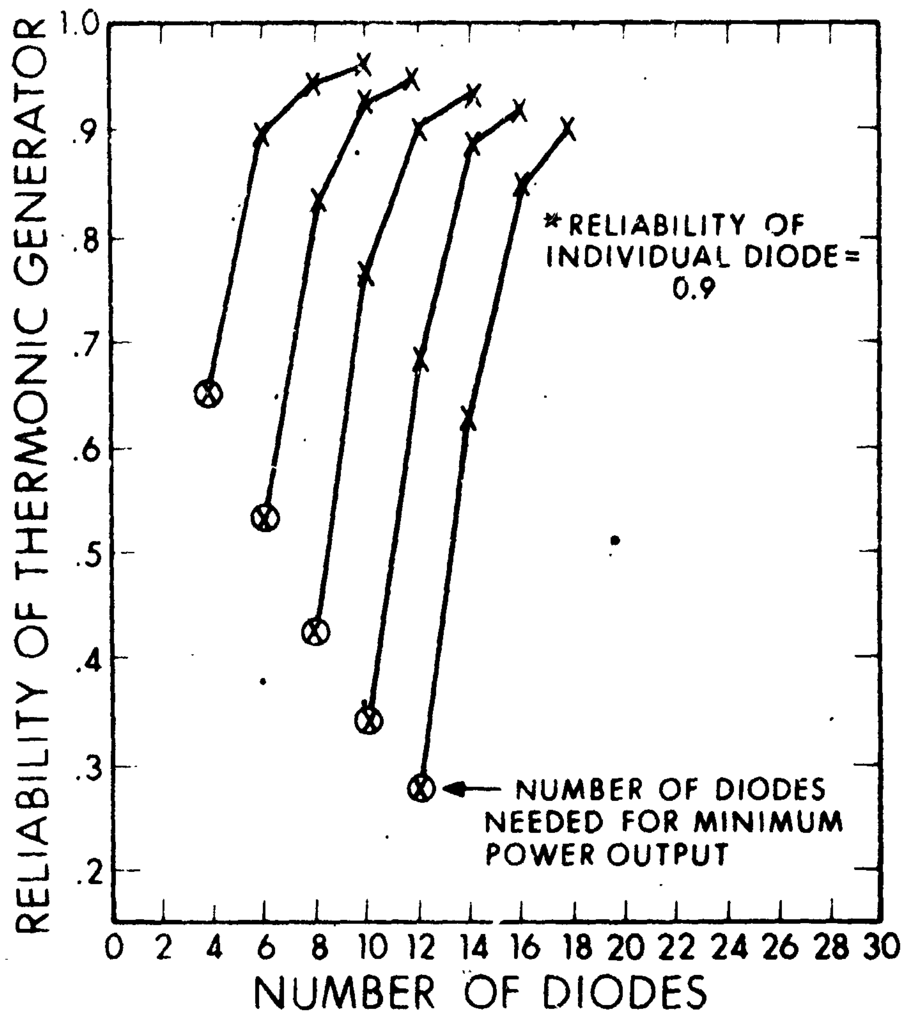
2. Three batteries sized such that any two of the three can fulfill all load requirements.
3. Extra source capacity. Ten percent was added to the photovoltaic array. For the thermionic generator, the number of extra diodes depended on the assumed diode reliability power level and circuitry attachment with the dc/dc converter. Sufficient redundancy was added to the thermicnic generator to achieve at least 0.9 reliability. Thermionic diode reliability of 0.9 was assumed in 1969 and 0.95 in 1975. Figure 2-3 illustrates typical reliability trends in thermionic generator and source analysis. Shown is a typical increase in generator reliability with increase in diodes, circuit connections to implement redundancy, and the reliability of the power source versus weight for several cases.

Based on assumptions described in detail in the appendices, reliability versus weight calculations were made for the missions, power levels and systems of interest. Table 2-III shows a typical weight and reliability analysis for redundant solar-thermionic systems which shows the increase in system component weight as the number of thermionic diodes is increased to improve system reliability. Particularly striking is the increase in dc/dc converter weight. Table 2-III shows a typical method of obtaining weight versus reliability estimates.

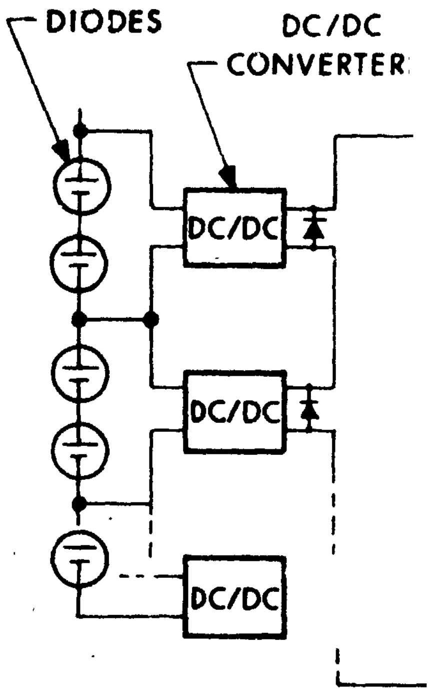
Figure 2-4 illustrates system component reliability versus weight numbers for three 1975 missions at the 1 KW level. Figure 2-5 shows the difference in weight for "redundant" and "nonredundant" systems for various missions at the 1 KW level.

These curves and tables are typical and illustrate several key points in considering the comparative reliability of the system.

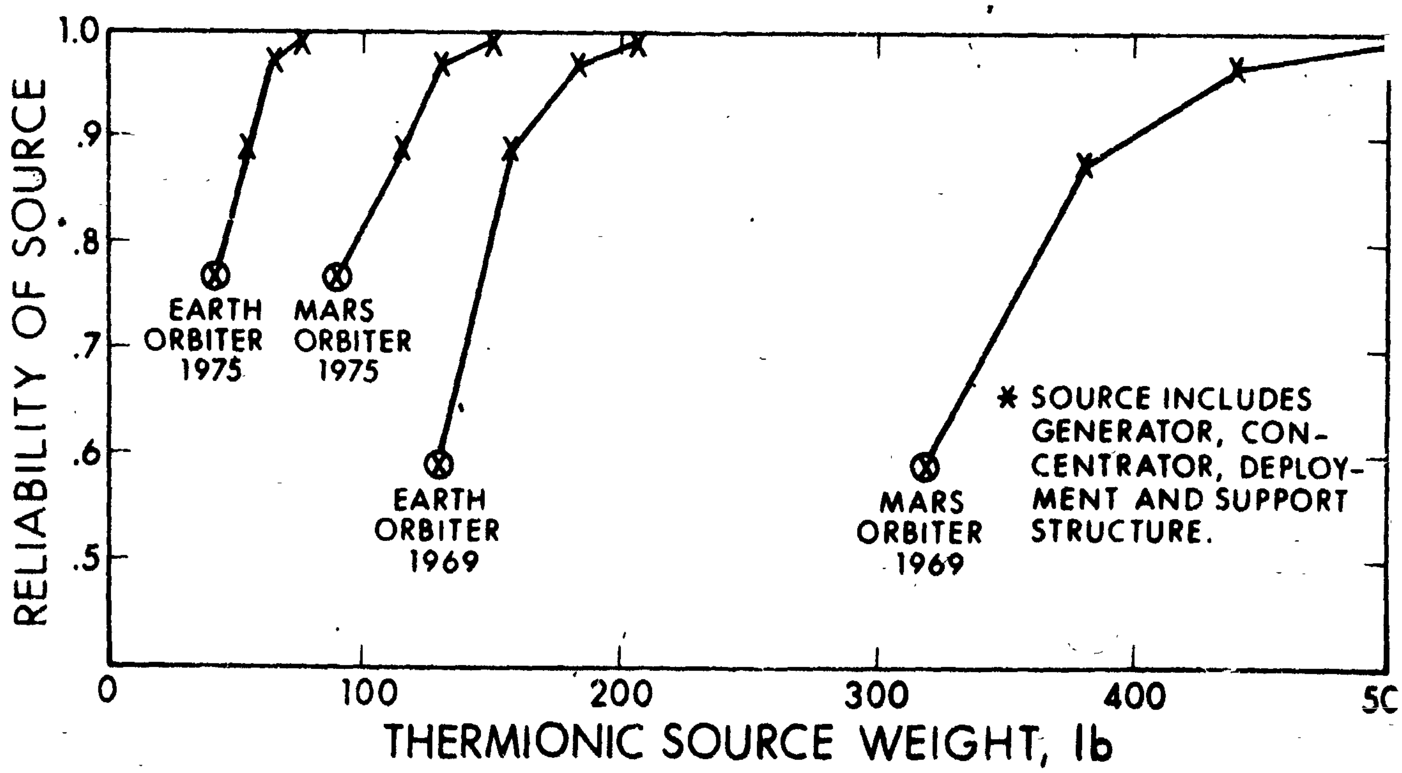
1. Power conditioning and battery subsystem reliability is close for either system. The power conditioning weight for solar-thermionics is generally heavier than for photovoltaics



(a) Reliability vs Number of Diodes Using Circuit in (b)



(b) Typical Circuit Used for Implementing redundancy in Thermionic Generator



(c) Typical Curves of Thermionic Source Reliability vs Weight

FIG. 2-3 TYPICAL RELIABILITY TRENDS FOR THERMIONIC SOURCES

Mission	Year	Load, Watts	F(1) Factor	Minimum Power Output from Source, watts	Number of Diodes in Generator(s)	Number of dc/dc Converters
Earth Orbiter 10,000 km	1969	200	1.715	343	7	1
					10	5
					12	6
		1000		1720	36	3
					60	15
	1975	200	1.715	343	7	1
					10	5
		1000		1720	36	3
					48	24
Solar Probe	1975	200	1.34	268	3	1
		1000	1.34	1340	27	2
					34	17
					40	20

(1) F factor refers to the ratio of raw-power from the source to conditioned power to the load; accounting for losses in power conditioning and energy storage; not including extra power from source due to redundant source elements.

TABLE

TYPICAL WEIGHT AND RELIABILITY ANALYSIS

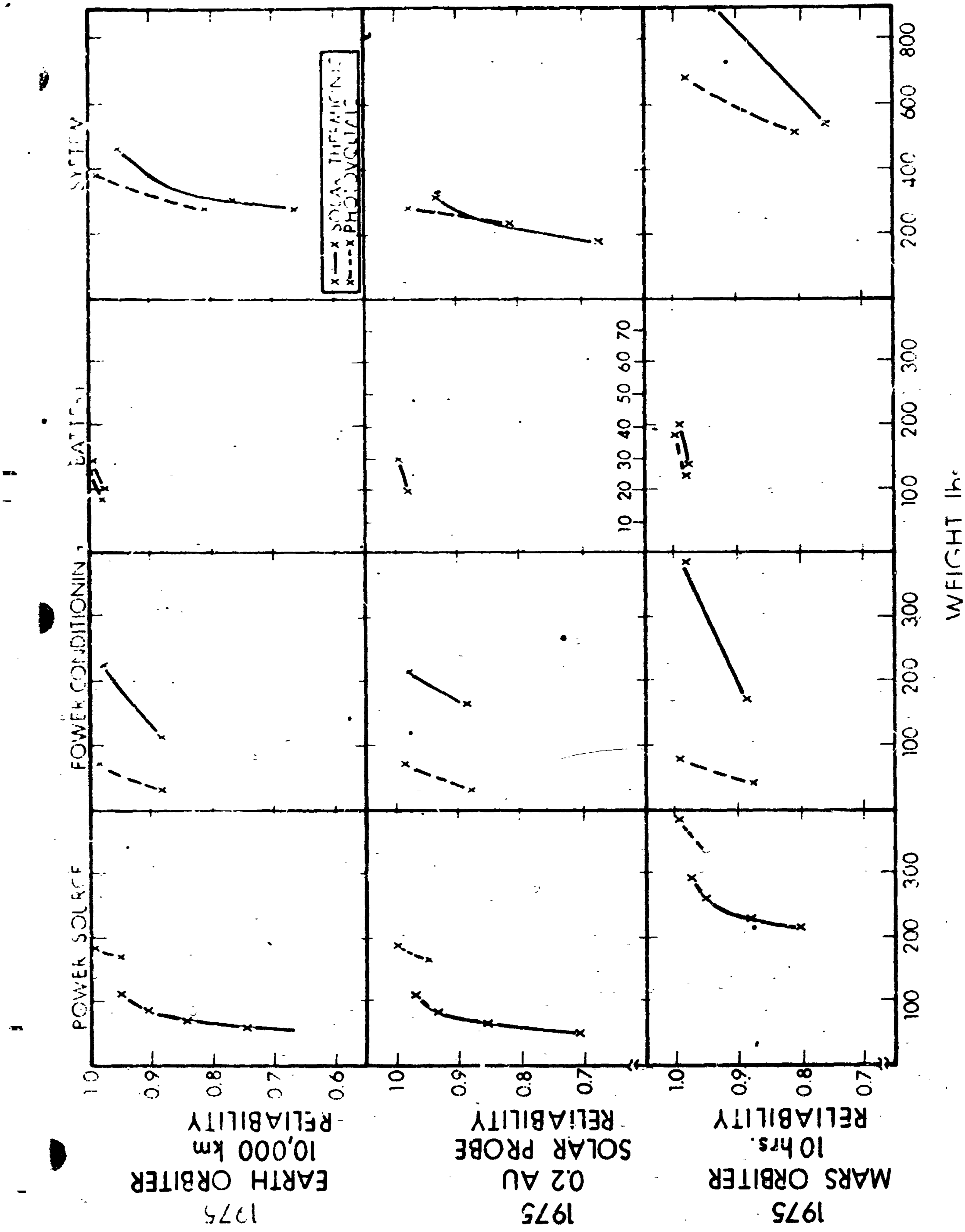
Source and dc/dc Converter Reliability	Nonredundant Source Weight, pounds	Redundant Source Weight, pounds	[illegible] Conv[illegible] watts
.47	37	--	343
.76		56	100
.905		68	100
~ 02	220	--	573
.862		360	200
.70	13	--	343
.98		19	100
~ .05	63		573
~ .92		87	100
.857	7	--	134
~ .05	48		620
.88		60	100
.97		70	100

2-III

FOR REDUNDANT SOLAR THERMIONIC SYSTEMS

...r,	dc/dc Converter Weight, pounds	Maximum Power Handled by Shunt Regulator, watts	Shunt Regulator Weight, pounds	Total Power Conditioning Weight, pounds	Battery Weight (Nonredundant) pounds	Battery Weight (Redundant) pounds	System Weight Without Redundancy, pounds	Redundant System Weight, pounds
	18	121	7	43	23	--	103	--
	55	256	11	84		34		174
	66	346	14	98		34		200
	72	540	17	145	115	--	467	--
	225	1520	24	305		172		837
	10	120	7	35	198		68	
	35	260	11	66		30		115
	52	540	17	117	99		280	
	3	1080	20	236		150		473
	8	--	--	28	2	--	37	
	25	--		78	21		147	
	99	300	12	165		[illegible]2		257
	140	580	17	210		32		312

(3)


1975
EARTH ORBITER
10,000 km
RELIABILITY
1975
0.2 AU
SOLAR PROBE
RELIABILITY
1975
MARS ORBITER
10 hrs.
RELIABILITY
POWER SOURCE
POWER CONDITIONIN
SOLAR THERMIONIC
PHOTOVOLTAIC
WEIGHT lb

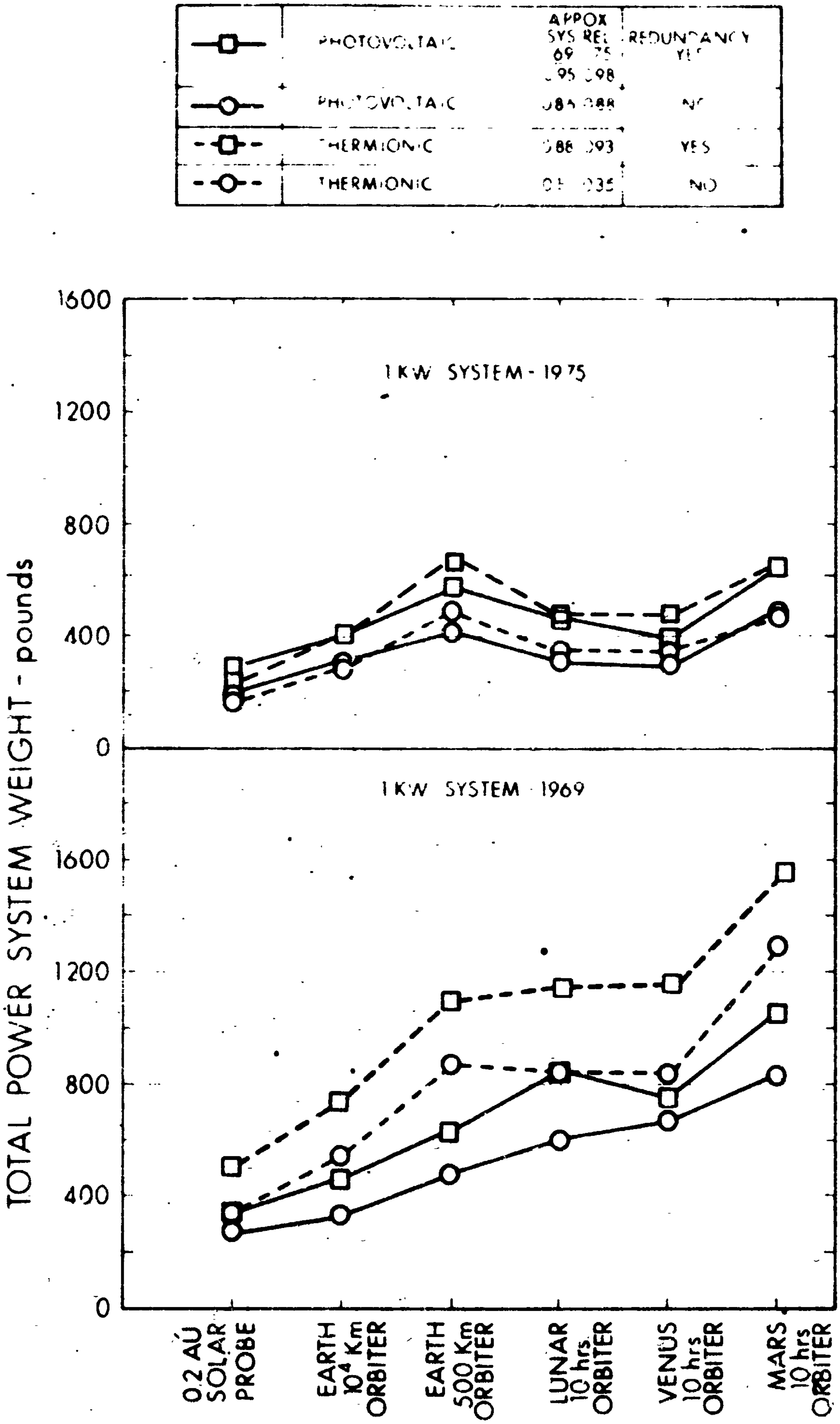


FIG. 2-5 TOTAL POWER SYSTEM WEIGHT FOR SELECTED MISSIONS AND SYSTEM RELIABILITY

and the weight penalty for reliability improvement is generally greater.

2. The key reliability problem of the solar-thermionic system is the demonstration of thermionic diode reliability. The generator power emanates from a relatively small number of sources (diodes). Examination of the number of test hours required to demonstrate reliability indicates probabilities of 0.95 to 0.99 could be demonstrated by 1971-3 depending on the intensity of the development effort. At the present time, it appears possible to develop thermionic diodes to highly reliable devices.

   The use of a relatively small number of diodes means that additional redundancy must be added to the generator in the form of properly connected diodes. The selected method of incorporating redundant thermionic diodes is shown in Fig. 2-3(b). Incorporation of redundancy increases the dc/dc converter weight significantly along with an increase in source weight.

3. The reliabilities of the solar-thermionic and photovoltaic systems are comparable if redundancy is added to the system. System reliability numbers of 0.9 to 0.98 can be anticipated for 1975 missions (see Table 2-III, Figs. 2-3, 4, and 5) and 0.85 to 0.95 for 1969 missions (0.85 was used as a lower limit for comparing 1969 system weights).

2.3 Weight

The solar-thermionic and photovoltaic systems have been compared on the basis of overall weight.

Power system weight versus conditioned power for both the photovoltaic and thermionic systems with and without redundancy are presented for every mission investigated during the program in the appendices. The packaging limitations were also investigated for each of the launch vehicles to be used, i.e., Atlas-Centaur and Saturn IB. For example, the Atlas-Centaur packaging limit for the thermionic system for a 10,000 km earth orbiter is 2700 watts in 1969.

The packaging limitation was obtained by a number of configuration studies based on reasonable estimates of spacecraft design. The maximum number of mirrors that can be packaged in the Saturn IB is fourteen 9.5-foot mirrors and the maximum number of concentrators that can be packaged in the Atlas-Centaur for maximum power output is eleven 7-foot-diameter mirrors. Using the maximum number of concentrators in conjunction with the concentrator diameter, along with the electrical power output as a function of mirror diameter, the maximum power capability of the launch vehicle can be ascertained.

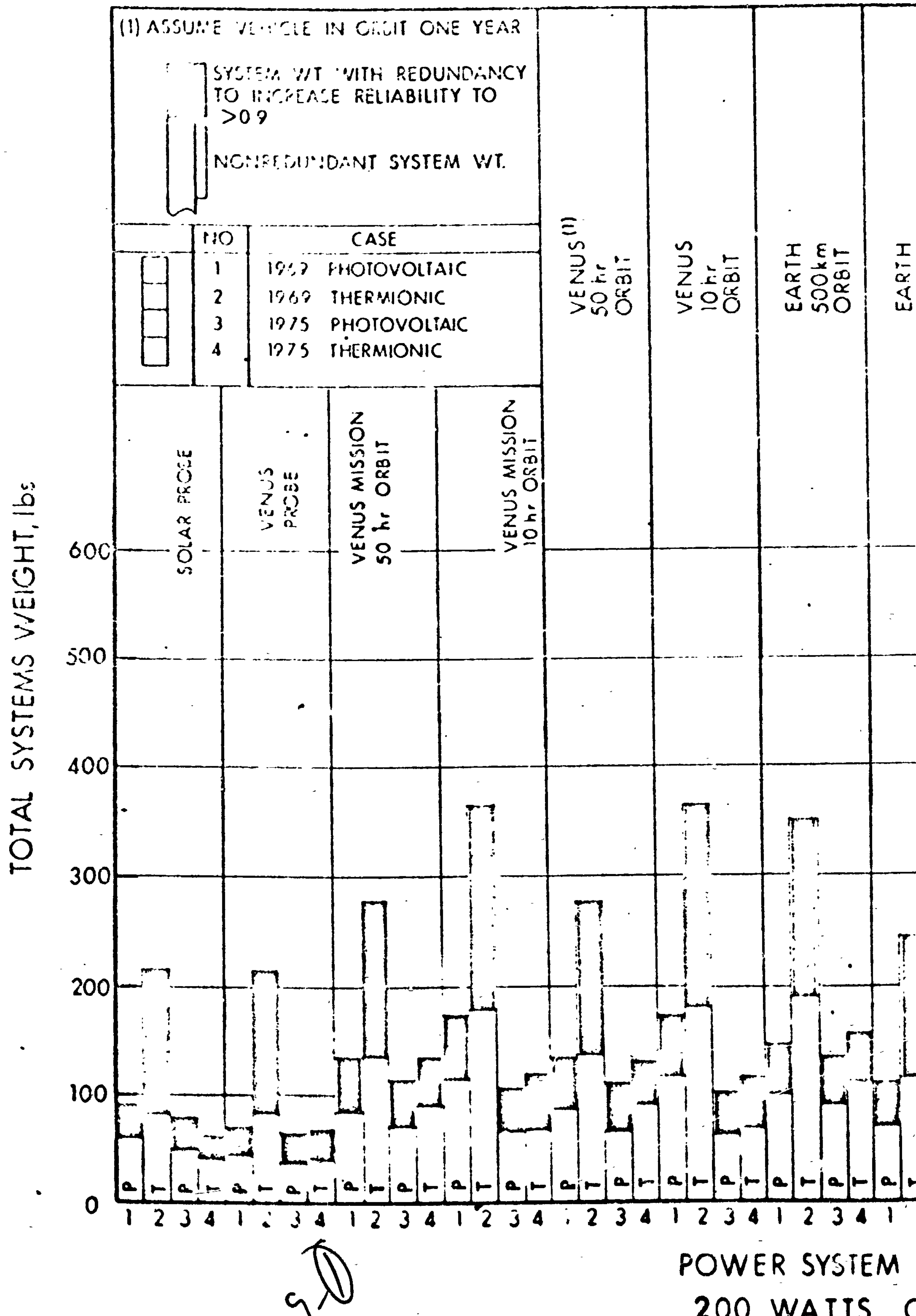
Figures 2-6, 2-7, and 2-8 summarize power system weight for a 200 W, 1 KW, and 4 KW conditioned power output for various missions. Shown are the following:

1. The weight of a nonredundant system for the years 1969 and 1975 (see previous discussion for reliability of systems).
2. The increased weight penalties to achieve a given amount of redundancy resulting in system reliability greater than 0.9.
3. Total system weight including redundancy.

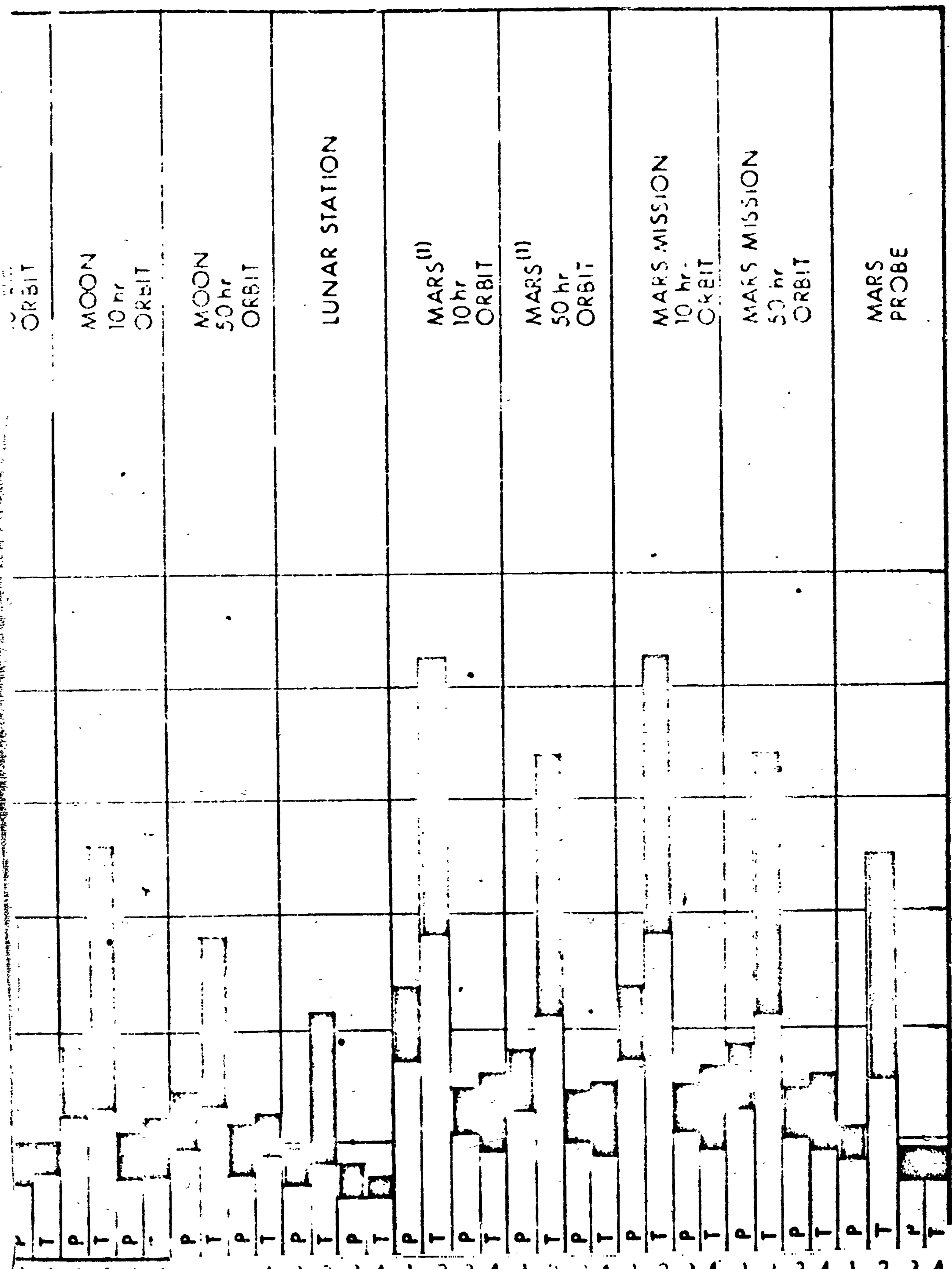
As shown by blank spaces in Fig. 2-8, it did not appear practical to package solar-thermionic arrays capable of providing sufficient power for a 4 KW system for several missions.

Based on the analysis conducted in this study, the following general conclusions can be drawn with regard to a comparison between solar-thermionic and photovoltaic systems.

1. The need for redundancy to increase solar-thermionic system reliability raises system weight significantly. It should be reemphasized, however, that reliability calculations are based on estimated diode reliabilities which can only be guessed at this time. Demonstration of thermionic diode reliability in the ball park of 0.97 to 0.99 would result in the situation where the solar-thermionic system was lighter in weight than photovoltaics for most missions in the year 1975.

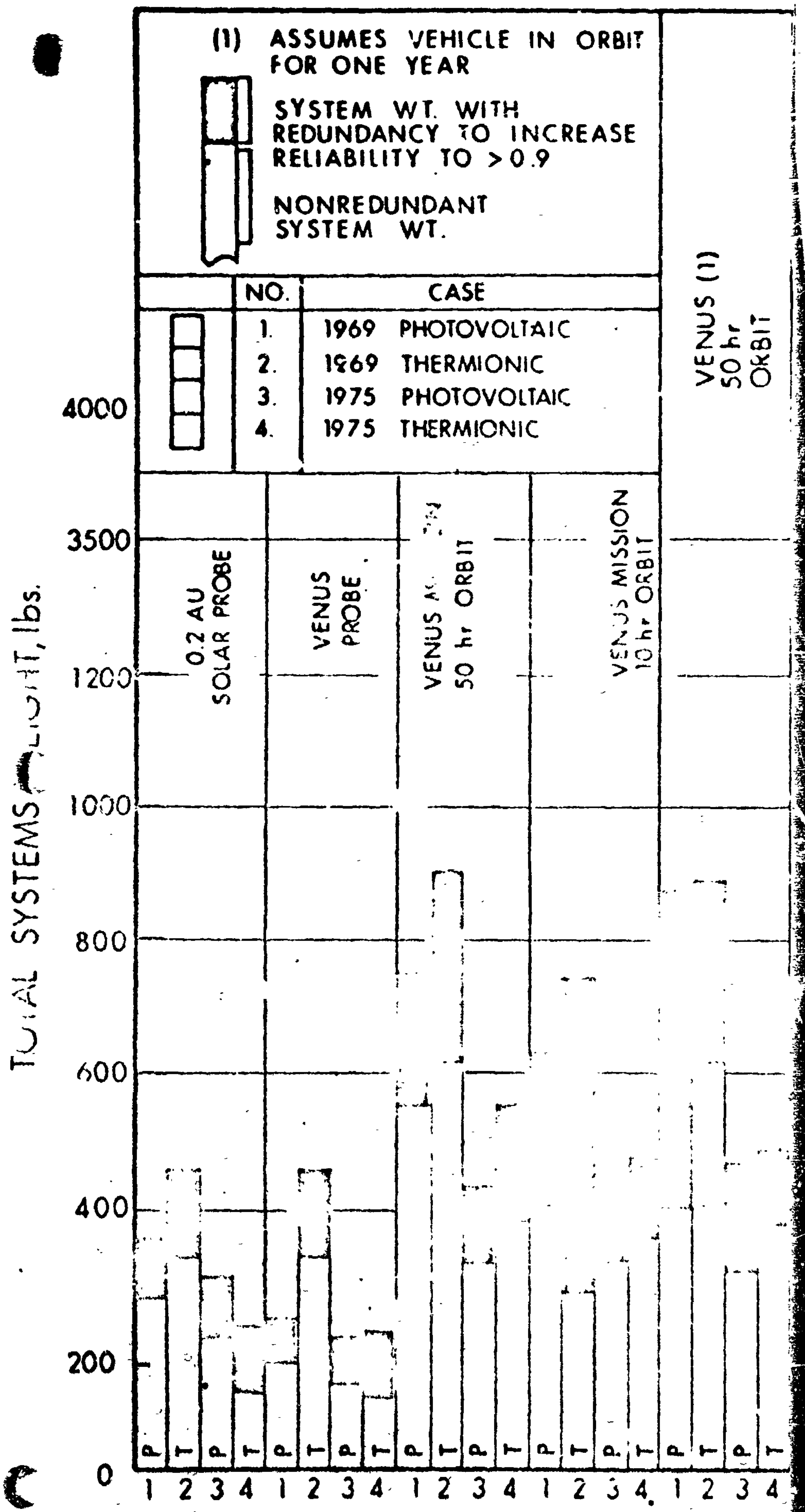

(1) ASSUME VEHICLE IN ORBIT ONE YEAR
SYSTEM WT WITH REDUNDANCY TO INCREASE RELIABILITY TO >0.9
NONREDUNDANT SYSTEM WT.
NO
CASE
1 1969 PHOTOVOLTAIC
2 1969 THERMIONIC
3 1975 PHOTOVOLTAIC
4 1975 THERMIONIC
SOLAR PROBE
VENUS PROBE
VENUS MISSION 50 hr ORBIT
VENUS MISSION 10 hr ORBIT
VENUS(1) 50 hr ORBIT
VENUS 10 hr ORBIT
EARTH 500km ORBIT
EARTH
TOTAL SYSTEMS WEIGHT, lbs
0
100
200
300
400
500
600
P
T
1 2 3 4
POWER SYSTEM
200 WATTS

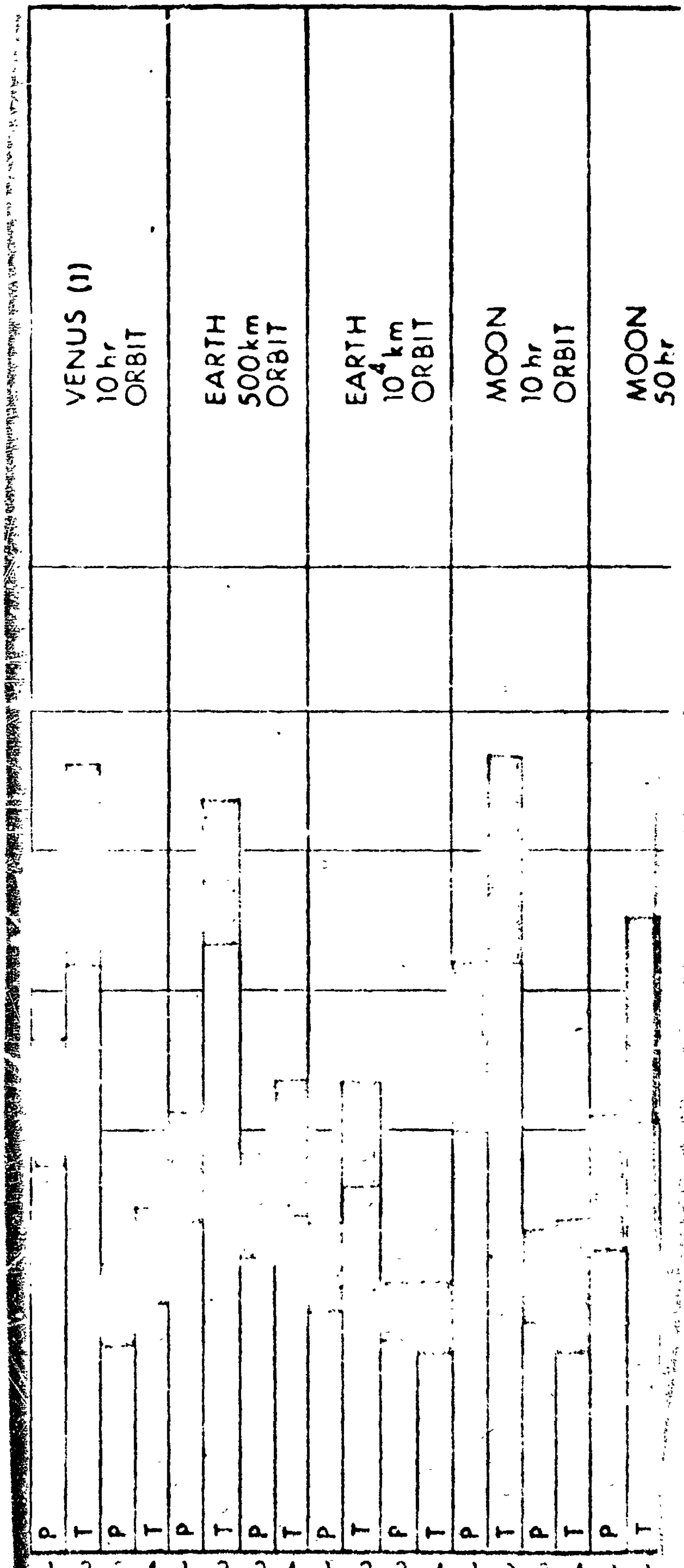
25-0



EIGHT COMPARISON
NDITIONED POWER

FIG. 2-6 POWER SYSTEM WEIGHT COMPARISON
200 WATTS CONDITIONED POWER


(1) ASSUMES VEHICLE IN ORBIT FOR ONE YEAR
SYSTEM WT. WITH REDUNDANCY TO INCREASE RELIABILITY TO >0.9
NONREDUNDANT SYSTEM WT.
NO.
CASE
1. 1969 PHOTOVOLTAIC
2. 1969 THERMIONIC
3. 1975 PHOTOVOLTAIC
4. 1975 THERMIONIC
0.2 AU SOLAR PROBE
VENUS PROBE
50 hr ORBIT
VENUS MISSION 10 hr ORBIT
VENUS (1) 50 hr ORBIT
TOTAL SYSTEMS
lbs.
4000
3500
1200
1000
800
600
400
200
0
P
T
1 2 3 4 1 2 3 4 1 2 3 4 1 2 3 4 1 2 3 4

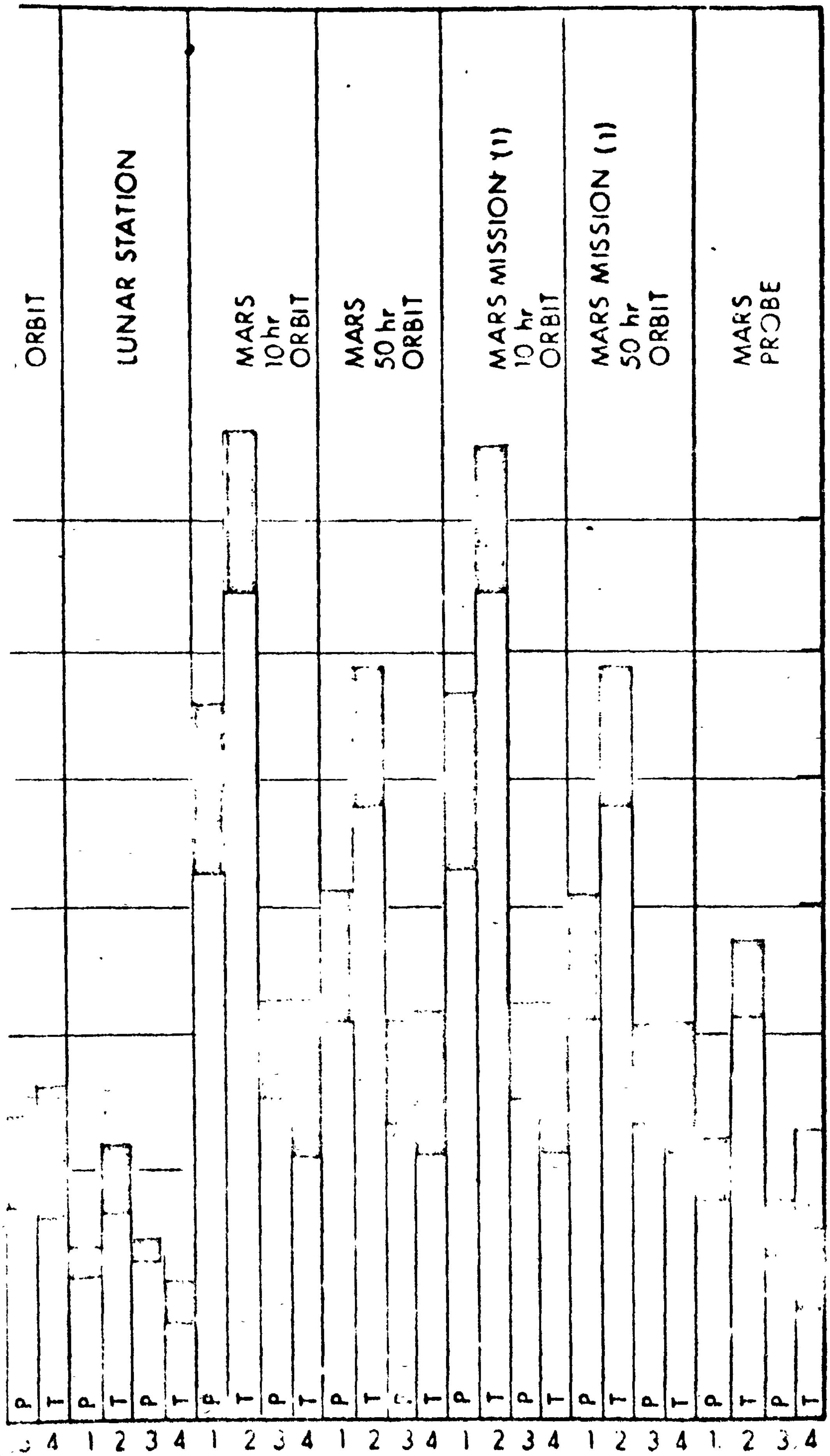


POWER SYSTEM WEIGHT COMPAR

1 KW CO'IDITIONED FO

2

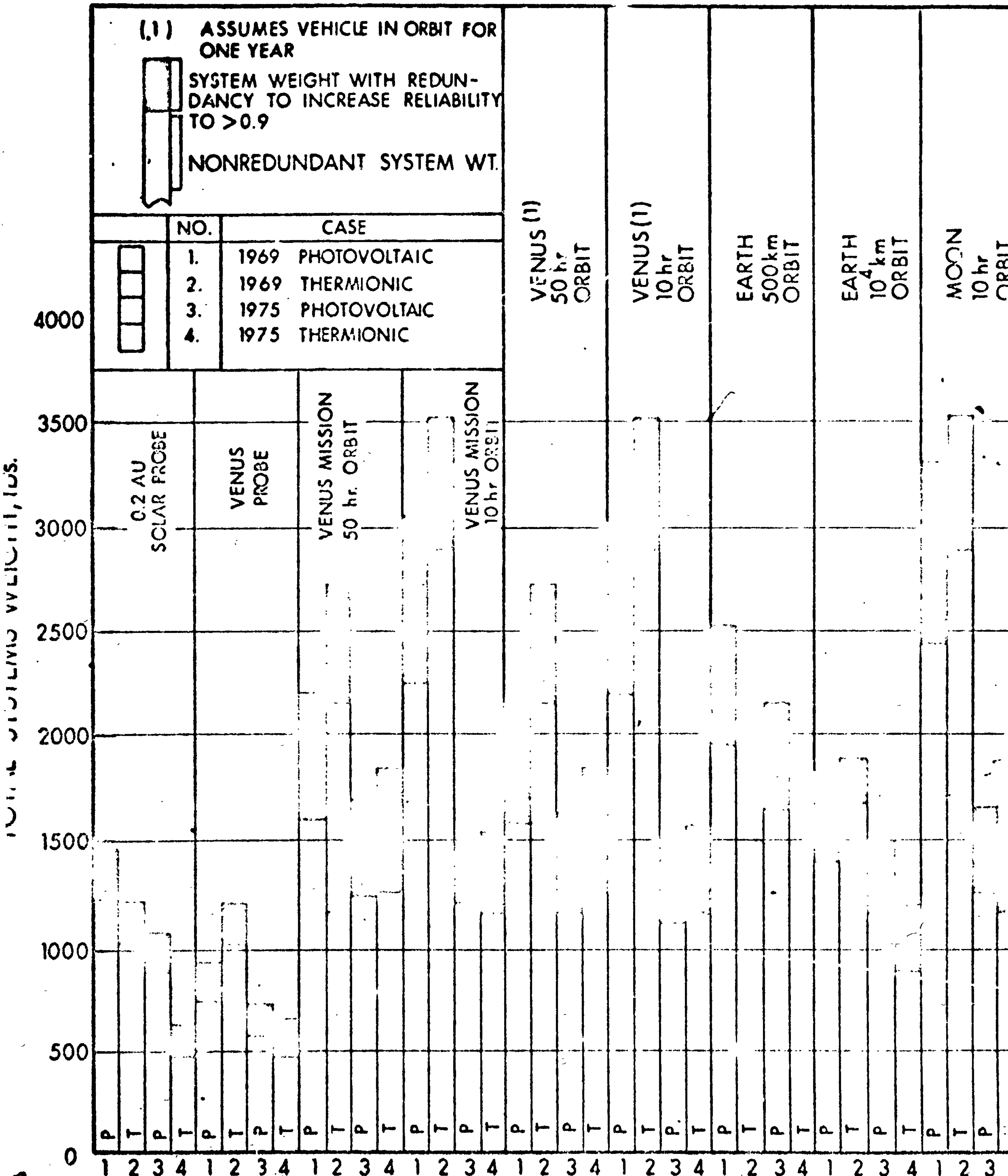
65-1049



RISON
WER

FIG. 2-7 POWER SYSTEM WEIGHT COMPARISON
1 kw CONDITIONED POWER

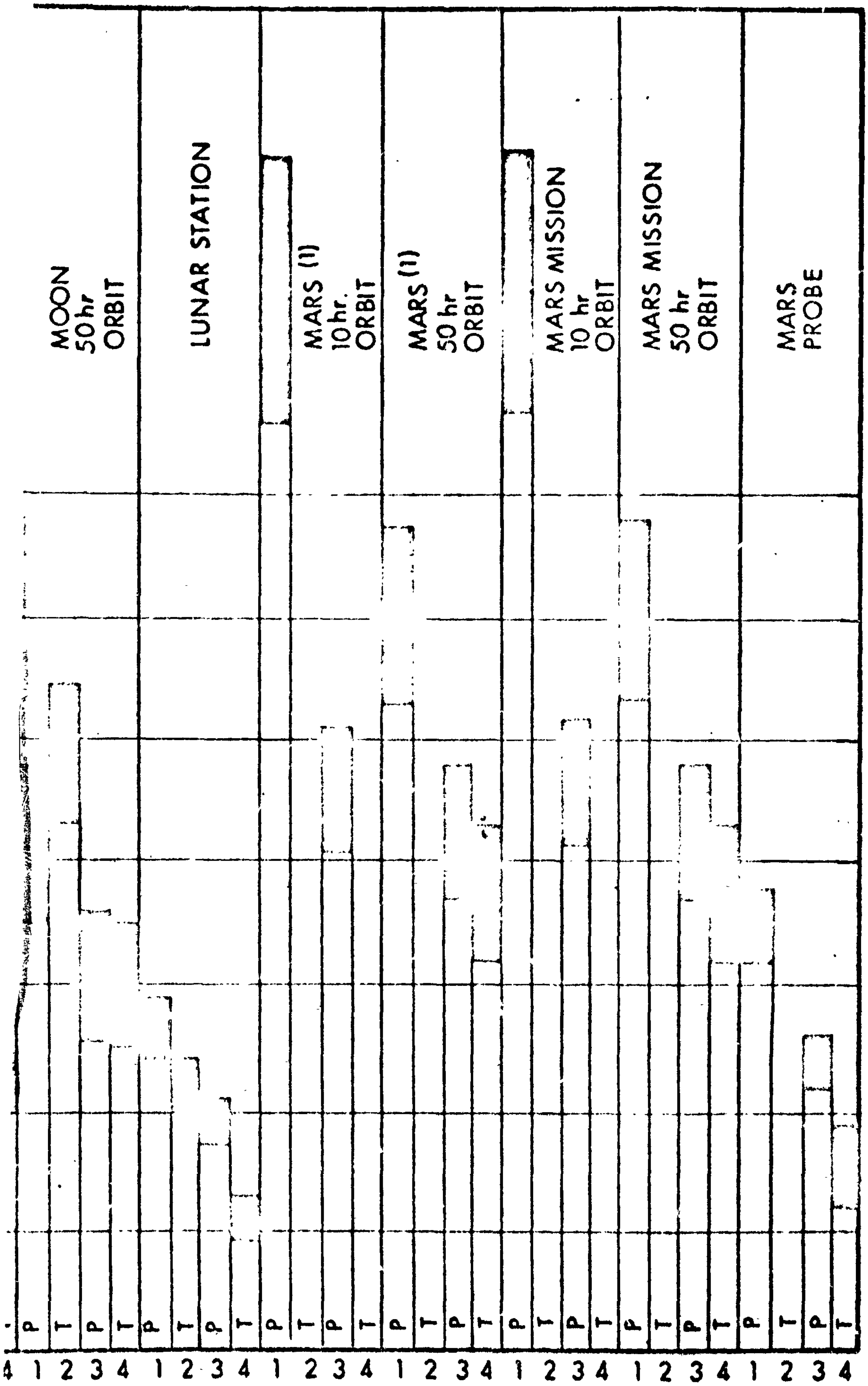
30 (3)



POWER SYSTEM WEIGHT CC
4000 WATTS CONDITIONE

31①

65-1049



MPARISON
D POWER

FIG. 2-8 POWER SYSTEM WEIGHT COMPARISON
4000 WATTS CONDITIONED POWER

2. The inclusion of attitude control system weight does not change the conclusion regarding weight comparison significantly except in the case of solar and Venus probes. In these cases, solar radiation pressure is a dominant perturbing torque and the use of solar-thermionics, with less surface area, results in attitude control weight savings.
3. All the system weights shown in Figs. 2-6, 2-7, and 2-8 include battery and power conditioning. In some cases, the weight of the power source is less than the weight of the battery plus power conditioning. For example, for a 0.2 AU solar probe (1975) the solar-thermionic source weighed 90 pounds, the power conditioning 210 pounds, and the battery 30 pounds. It is important to examine the comparison of power sources (photovoltaic or thermionic) separate from power conditioning and batteries. This is because power conditioning and battery weight are subject to a considerable alteration as technology develops. Furthermore, system weights shown in Figs. 2-6, 2-7, and 2-8 are based on the use of a battery with only 30 percent depth of discharge. For the redundant case, two out of three batteries can handle the load. In this sense, battery weights are extremely conservative.
4. The approach toward structural analysis was conservative. Structural criteria such as minimum resonant frequencies of 2 cps and the ability to withstand a 3 g maneuver introduced a note of conservatism in the analysis. Thus, "ultralightweight structures" were not considered, as the use of this type of structure is dependent upon modifications in vehicle mission and design which were not considered to be part of this study.
5. The available watt/sq ft figure for solar-thermionics is generally higher than that of photovoltaics in 1975.

However, the three-dimensional feature of a thermionic system presents more of a problem in view factors with items such as scanning platforms, antennas, and instrumentation.

6. The analysis conducted assumes that a constant power level must be available from each system from initiation to completion of mission. For the photovoltaic system, this means the array is sized by the Mars solar intensity for a Mars mission, and the earth's solar intensity for all other missions with the exception of the solar probe. For the solar-thermionic system, system size is dictated by the Mars solar intensity for all Mars missions and the earth's solar intensity for all other missions. This type of comparison is, in a sense, unfair to thermionics since a large amount of the available energy to the solar-thermionic system on a Venus or a sun probe must be wasted because of the need to size for earth conditions.

   It is felt that further study should be made wherein the vehicle load demand increases as the vehicle approaches the sun, with a minimum power demand near earth. Under these conditions, preliminary analysis shows that the weight advantage of the solar-thermionic system becomes quite significant.

7. In all analyses, solar panel temperature was allowed to rise to a fairly high level. For the 0.2 AU solar probe case, panel temperatures approaching 150°C were reached (close to the solder melting point). The long-term reliability of the solar panel at these temperatures probably can be made high, but further work is required to verify the long-term effects of high intensity and high temperature on solar panel performance. In a similar manner, all analysis assumes a solar concentrator reflectivity of 0.85 with no degradation. The increase in solar thermionic source weight would be roughly inversely proportional to a decrease in reflectivity if degradation occurs. Possible degradation mechanisms are vapor deposition from the hot generator and solar proton damage; these effects must be known prior to solar-thermionic system flight.

3. AREAS REQUIRING FURTHER DEFINITION

The study concludes that solar-thermionic systems can compete with photovoltaics in the early 70's, with predicted component developments. Study conclusions are based upon judgments concerning available component state of the art, available mechanization and packaging techniques, and assumed models for reliability of the system explained in this report.

During the course of the comparison study, it was apparent that in several areas the amount of knowledge was limited and further development work was necessary to better define the system. Reasonable judgments were made in each of these areas; however, it is useful to outline some of the major problem areas:

<u>Photovoltaic</u>

1) Structure design for high power, large area arrays
2) Solar cell performance at high solar intensities close to the sun
3) High power level power conditioning design
4) Definition of battery limits or depth of discharge, cycle life, and other parameters

<u>Solar-Thermionic</u>

1) Solar concentrator reflectivity and structure design
2) Deployment structure alignment
3) Generator reliability demonstration and definition of limits on necessary current regulation
4) Decrease in power conditioning weight and high power level design
5) Definition of operational constraints on Thermal Energy Storage/diode unit

6) Design and test of passive flux control

7) Definition of ground test

8) Battery definition

In the photovoltaic system, major problem areas include:

1. Definition of the structure. The design of the solar panel structure is strongly influenced by the vehicle, mission, and environment. Rigid, flat structures, fixed to the vehicle body in a manner similar to Mariner IV, are suitable for power levels up to 1 - 2 kw. Beyond this range, structural weights begin to increase to high levels. Other structural techniques such as retractable structures, etc., should be investigated in conjunction with alteration of vehicle requirements to lessen maneuver loads, lower or eliminate resonant frequency criteria, minimize boost loads through structural damping and other techniques. The behavior of solar panel structures at environmental extremes such as at 0.2 AU must be carefully investigated. For example, a solar panel structure temporarily shielded from an intense sun is likely to experience severe thermal shock problems. It must be emphasized, therefore, that generalized curves of weight and area such as presented in this study, are only approximations which must be verified by detailed structural design for each specific case.
2. Solar cell performance. Within the mission requirements between Venus and Mars, solar cell performance has been adequately defined in laboratory experiments and through actual use. For solar probes, however, solar cell behavior is not well defined. For example, at 0.2 AU, performance can vary by as much as a factor of 3, depending on assumptions regarding internal series resistance of the cell. The behavior of solar panels close to the sun is critically

dependent on detailed knowledge of solar cell performance and this area must be further investigated.

As an example, at 0.2 AU, assuming a solar cell series resistance of 0.4 ohms, the solar panel must be inclined at an 82° angle to the sun and a deviation of 3 degrees can decrease power by 20 percent. A small change in cell performance and series resistance can drastically change the optimum panel inclination.

3. Power conditioning for high power levels. The design and operation of the power conditioning for the photovoltaic system is fairly well defined at lower power levels (less than 1 kw). Furthermore, redundancy can be added at little cost in weight. At high power levels, however, the choice of power conditioning schemes is not clear cut. For example, it becomes advantageous in some instances to convert the raw power dc directly to ac on the solar panel. Also, solid state component performance limits are such that a large number of redundant converters and regulators are required to handle the entire power load. Problems of heat dissipation and ground loops become more severe also.

4. Battery definition. The design of the energy storage system is hampered by lack of knowledge of battery reliability under varying conditions of depth of discharge, cycle conditions, etc. The battery design in this study is conservative (e.g., 30 percent depth of discharge limit) and battery weight could be considerably lessened with additional knowledge.

In the solar-thermionic system area, the following major problem areas should be attacked as soon as possible.

1. Solar concentrator. A principal solar concentrator problem area is the definition of reflective coating degradation in space. Factors needed to define reflectance with time include dependence on temperature, dependence on meteoroid and solar protons, dependence on uv, and related factors. A second area which requires further analysis is the ability of the concentrator skin to withstand vibration and shock loads. At present, material properties of the nonmagnetic electroformed materials can only be approximated, and the properties of large paraboloid single skins are not well understood.
2. System deployment structures. Design of the deployment structures entails the same type of problem as solar panel deployment structures in that interaction with the vehicle and mission is critically important. An additional problem, and perhaps resulting in an increase in weight, is the need for accurate alignment of the entire deployment structure and/or the inclusion of a vernier orientation scheme to maintain tight orientation accuracies.
3. Generator. With the assumption that the thermionic diode converter performance will continue to improve at the rapid pace experienced over the past few years, the principal problem area in generator design is the definition of required

diode reliability, the demonstration of diode reliability, and the method by which redundancy is incorporated. The power output in the system comes from a small number of thermionic diodes. Therefore, the reliability of each diode must be high and the incorporation of redundancy techniques or standby generators must be carefully engineered. Associated with this problem is the definition of the most likely mode of failure of the diode and the after effects (both thermal and electrical) on other diodes. Reliability can be incorporated into the generator by extra diodes suitably electrically connected (at the cost of increased system weight and mirror size), paralleling of diodes (at the cost of decreased system efficiency), and the use of additional dc/dc converters for isolating diode circuits.

Diode reliability and life data must be obtained in the near future.

In addition, sufficient performance data concerning the operation of the diode under constant power input is lacking. It is not clear what the limitations on load changes are, nor what temperatures will be encountered in the diode structure with a load change. Data of this type must be obtained for valid system design.

4. Power conditioning. The power conditioning for the solar-thermionic system is perhaps twice the weight of the photovoltaic system for the same load power level. A principal requirement of the power conditioning for the solar-thermionic system is the need to provide constant current regulation to the generator. This requirement is imposed by the need for the thermionic diode to operate near its design point in order that diode temperatures may be predicted. The design of the shunt load required for generator regulation is highly dependent upon the amount of redundancy incorporated into the generator, the amount of regulation desired, and the expected

variation in generator output. Shunt load weight and heat load dissipation problems can be critical. The need for a shunt load may be alleviated by the use of heat pipe technology in the generator. In addition to the shunt load, the selection and design of the dc/dc converter is another serious problem. The number of dc/dc converters, the effect on reliability, the need to isolate electrically a certain number of diodes, and other factors must be considered in this design.

At present, the dc/dc converter weight accounts for about half of the power conditioning weight in a redundant system. New circuitry and devices must be devised to cut the dc/dc converter weight.

5. Thermal energy storage. The incorporation of thermal energy storage into the generator will require careful study. Thermal energy storage was investigated and, using assumptions regarding component technology disucssed in the appendices, resulted in decreased system weight for the earth orbiter cases. The operational constraints and performance of the diode/TES unit are relatively unknown, and must be determined before any conclusions can be made.
6. Solar flux control. Preliminary design indicates that a solar flux control unit will be useful in those cases where a TES unit is used and for solar probe missions. In the case of the use of TES, the solar flux control could serve to compensate for variable darktimes. For solar probes, a solar flux control will be needed to compensate for the excess amount of energy near the sun. It appears possible in the solar probe case to use an entirely passive flux control but tests must be devised to verify this.
7. Ground test. A principal problem in the use of solar-thermionics is the ability to perform flight qualification tests which are valid. The need for vacuum conditions for the generator and the need for larger mirrors or auxiliary heaters indicates the need for special test facilities and a carefully laid-out program to insure that ground tests truly reflect space performance.

4. SOLAR-THERMIONIC SYSTEM DESCRIPTION

4.1 Component Performance

The preferred power conditioning mechanization for the solar-thermionic system is shown in Fig. 4-1 with the addition of redundant modules to increase reliability. The system consists of the generator and its dc/dc converters, a parasitic load which in itself consists of a shunt load and a battery charger, batteries, battery converters, voltage regulator (series switching regulator), and dc/ac inverters.

The series switching regulator, dc/ac inverter and battery converter each have an identical unit in the standby mode with a failure sense and switching unit that will detect a failure of the primary unit and initiate switching to the standby unit.

For the purposes of this study, three batteries are employed in an operational redundant configuration where they are sized such that any two of the batteries can carry the normal system load. The shunt loads are similarly sized so that in the event of a failure of one, the remaining two can adequately handle the total power to be dissipated. The diodes of a thermionic generator system should, if possible, be electrically connected in series to minimize losses. Furthermore, the load current of those diodes should be held relatively constant to regulate diode temperatures. The power conditioning equipment is designed to provide a constant current load on the thermionic generator during the normal sunlight operation regardless of the load power. During sunlight operation, excess power is consumed by the parasitic shunt load or in charging the batteries. Other modes of operation are explained in the Appendices.

Battery development schedule is shown in Table 4-I; this information was used as ground rules for the analysis.

Estimates of generator efficiency as a function of cavity temperature and year are shown in Table 4-II. This efficiency was derived from projected converter efficiencies provided by JPL. It appears

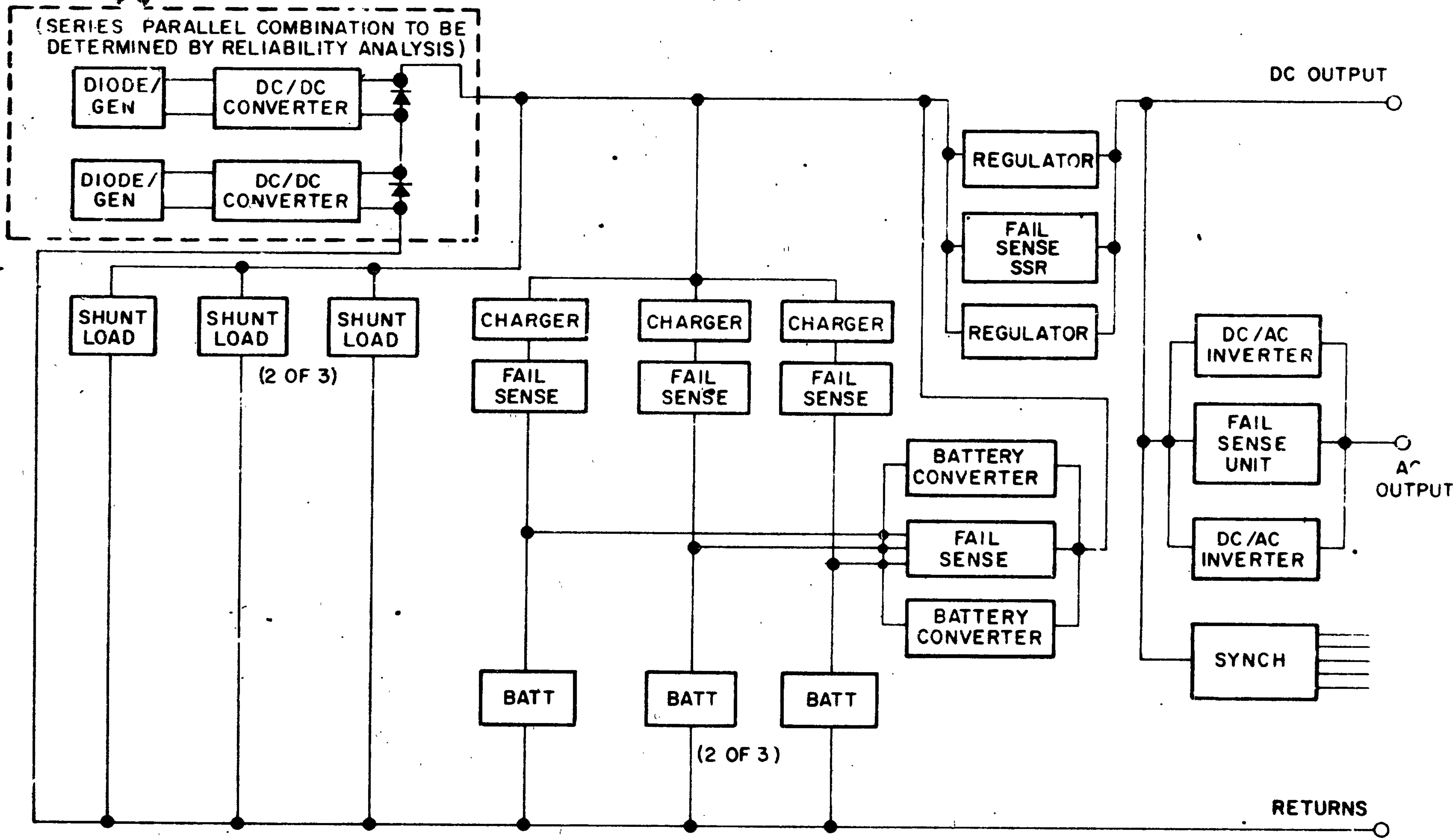


FIG. 4-1 SOLAR-THERMIONIC BLOCK DIAGRAM WITH REDUNDANT MODULES

TABLE 4-I

BATTERY DEVELOPMENT SCHEDULE

	1969	1971	1973	1975
SECONDARY BATTERIES				
Sealed Ag-Zn				
95% capacity retention at 70°F	1 yr	1½ yr	1½	2 yr
Watt-hours per pound	60	70	75	80
Storage life	2 yr	2½ yr	3 yr	4 yr
Watt-hours per cubic inch	2.0	2.1	2.2	2.3
Cycles before failure at 30% depth of discharge	500	700	1000	1500
Operating temperature range (internal)	-60° to 160°F	-60° to 200°F	-60° to 250°F	-60° to 300°F
Sealed Ag-Cd				
95% capacity retention at 70°F	1 yr	1½ yr	2 yr	2½ yr
Watt-hours per pound	30	32	34	35
Storage life	2 yr	3 yr	4 yr	5 yr
Watt-hours per cubic inch	1.69	1.75	1.78	1.81
Cycles before failure at 30% depth of discharge	2000	3000	4000	5000
Operating temperature range (internal)	-40° to 165°F	-40° to 200°F	-40° to 250°F	-40° to 300°F
Sealed Ni-Cd				
Watt-hours per pound	15	15	16	17
Storage life	5 yr	5 yr	6 yr	6 yr
Watt-hours per cubic inch	0.50	0.50	0.53	0.55
Cycle life at 30% depth of discharge	10,000	12,000	14,000	15,000
Operating temperature range (internal)	0-200°F	0-250°F	0-300°F	0-300°F

TABLE 4-II

GENERATOR EFFICIENCY

Temperature	1969	1970	1971	1972	1973	1974	1975	1976	1977
1400°C	7.5	11.8	11.8	12.7	12.7	15.9	15.9	16.9	16.9
1500°C	8.6	13.4	13.4	14.5	14.5	18.1	18.1	19.2	19.2
1600°C	9.4	14.6	14.6	15.8	15.8	19.8	19.8	20.9	20.9
1700°C	10	15.6	15.6	16.9	16.9	21.1	21.1	22.4	22.4
1800°C	10.4	16.3	16.3	17.6	17.6	22	22	23.3	23.3
1900°C	10.6	16.6	16.6	17.9	17.9	22.4	22.4	23.7	23.7

reasonable in light of recent developments. More details of generator efficiency calculations are contained in the Appendixes.

Weights of individual converters able to provide 50 watts each range from 0.6 lbs in 1969 to 0.25 lbs in 1977. It was assumed that the generator structure weight is about equal to the combined weight of the converters. It was also assumed that twenty-diode generators are possible for high power levels.

For the solar probe, the weight of the solar flux control system was ignored as this will generally be a very small percentage of system weight. It is recommended, after examination of generator performance, that a radiation dissipative shield be included on the front of the generator to enable the incorporation of a passive flux control for most missions.

Examination was made of the potentiality of using thermal energy storage to minimize the need for storage batteries. A typical conclusion is shown in Fig. 4-2 which illustrates system total weight as a function of storage dark time using 1975 component performance. As shown, thermal energy storage appears to decrease system weight for the earth orbiting missions. Also, the difference between 1500°C and 1700°C operation is small and 1500°C operation may be advantageous from a reliability viewpoint. Assumptions regarding the nature of thermal energy storage, materials used, structural, and storage efficiencies are discussed in the Appendixes. One potential advantage of TES (not considered in the study) is the minimization of battery requirements to provide power during midcourse maneuvers.

A detailed analysis was made of the concentrator/cavity performance. Computer calculations were made assuming variation in a large number of parameters and typical results are shown in Figs. 4-3 through 4-6.

Figure 4-3 shows the specific weight of the concentrator (lbs/sq ft). Specific weight is expected to decrease with decreasing mirror diameter which can be expected from an examination of the structural characteristics of the single-skin electroformed concentrator.

Collector absorber efficiency does not change significantly with mirror diameter and was assumed to be approximately constant in the

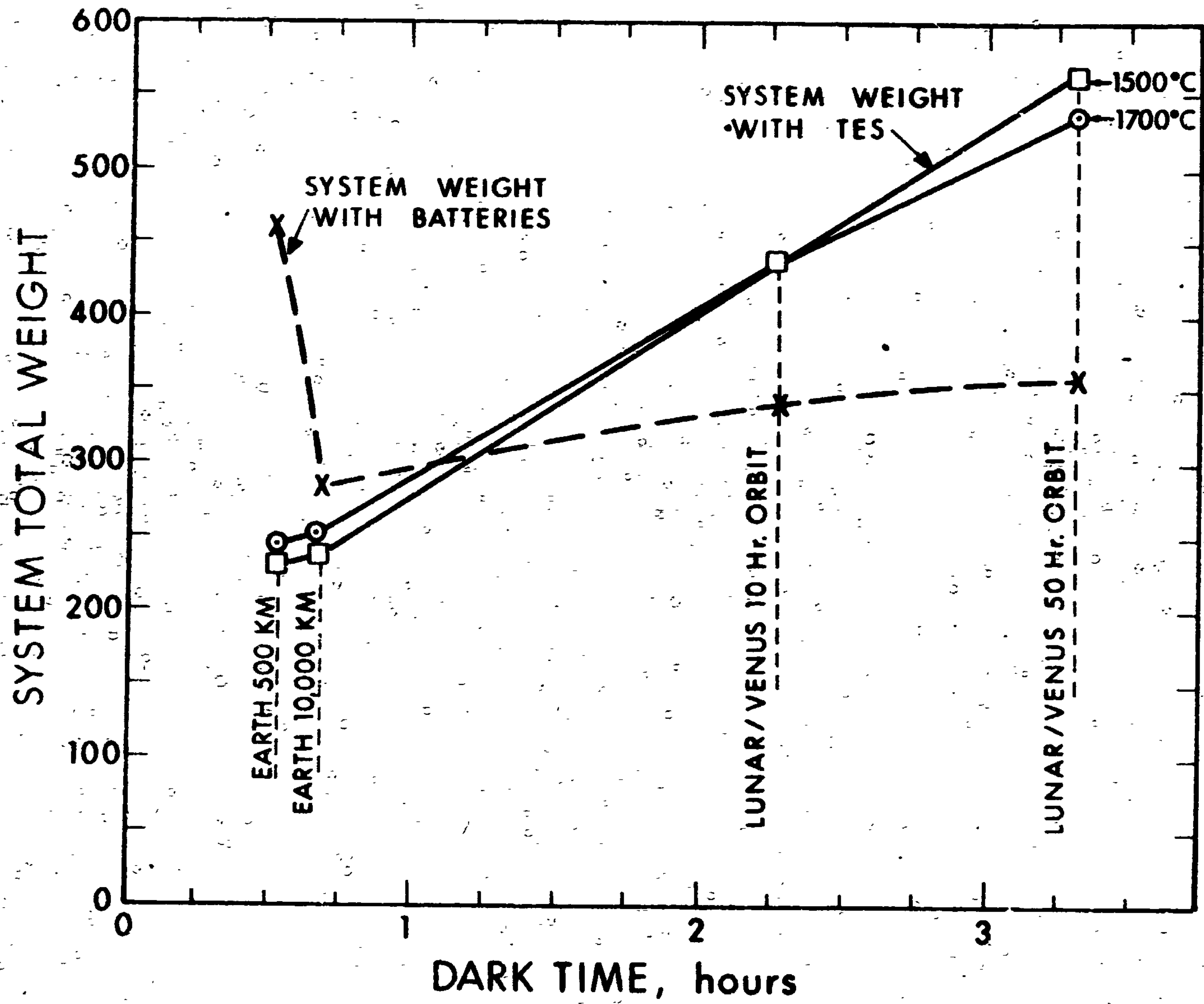


FIG. 4-2 COMPARISON OF SYSTEM WEIGHTS USING BATTERIES OR TES STORAGE — 1 KW POWER SYSTEM (1975)

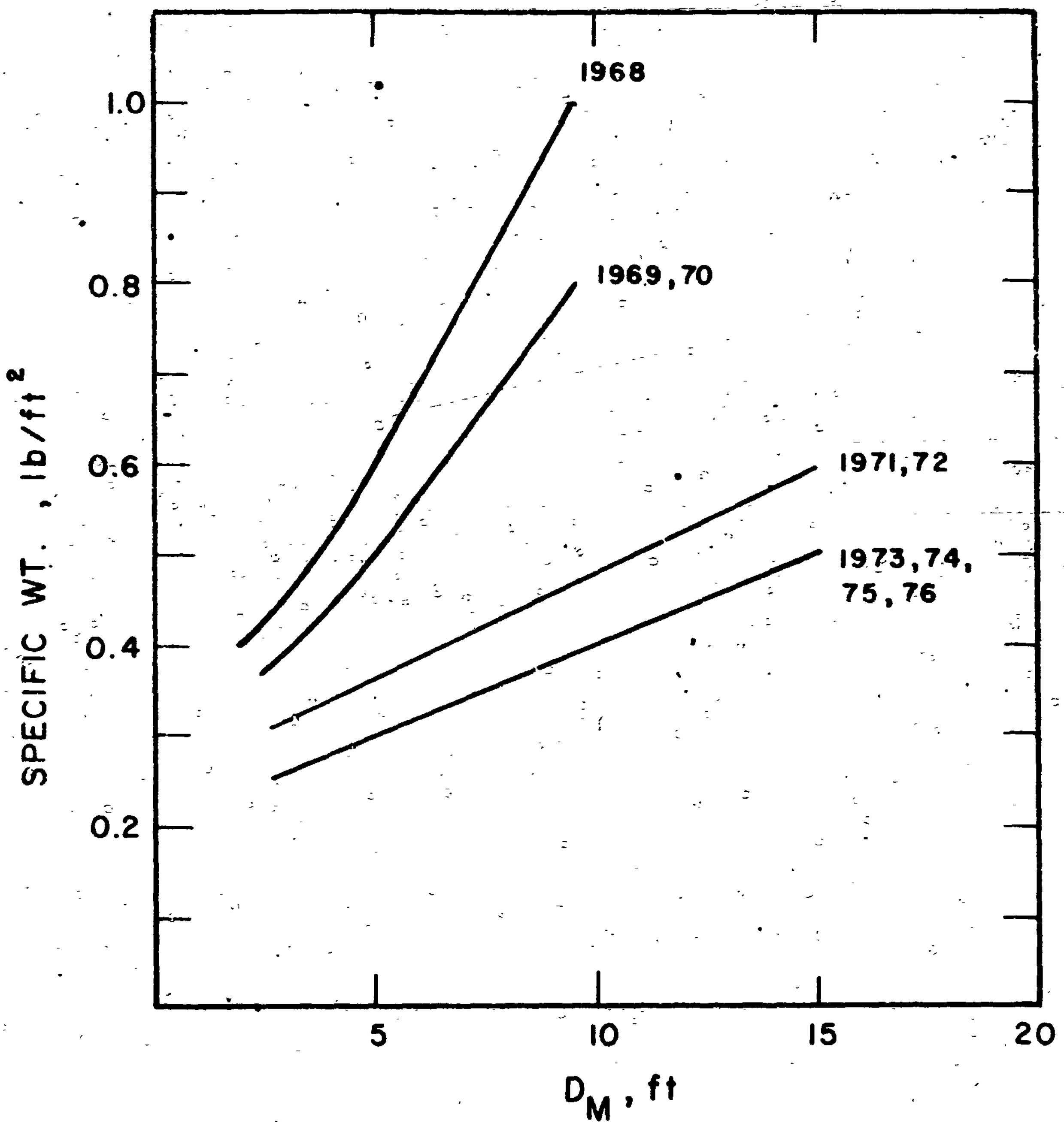


FIG. 4-3 SOLAR CONCENTRATOR SPECIFIC WEIGHT

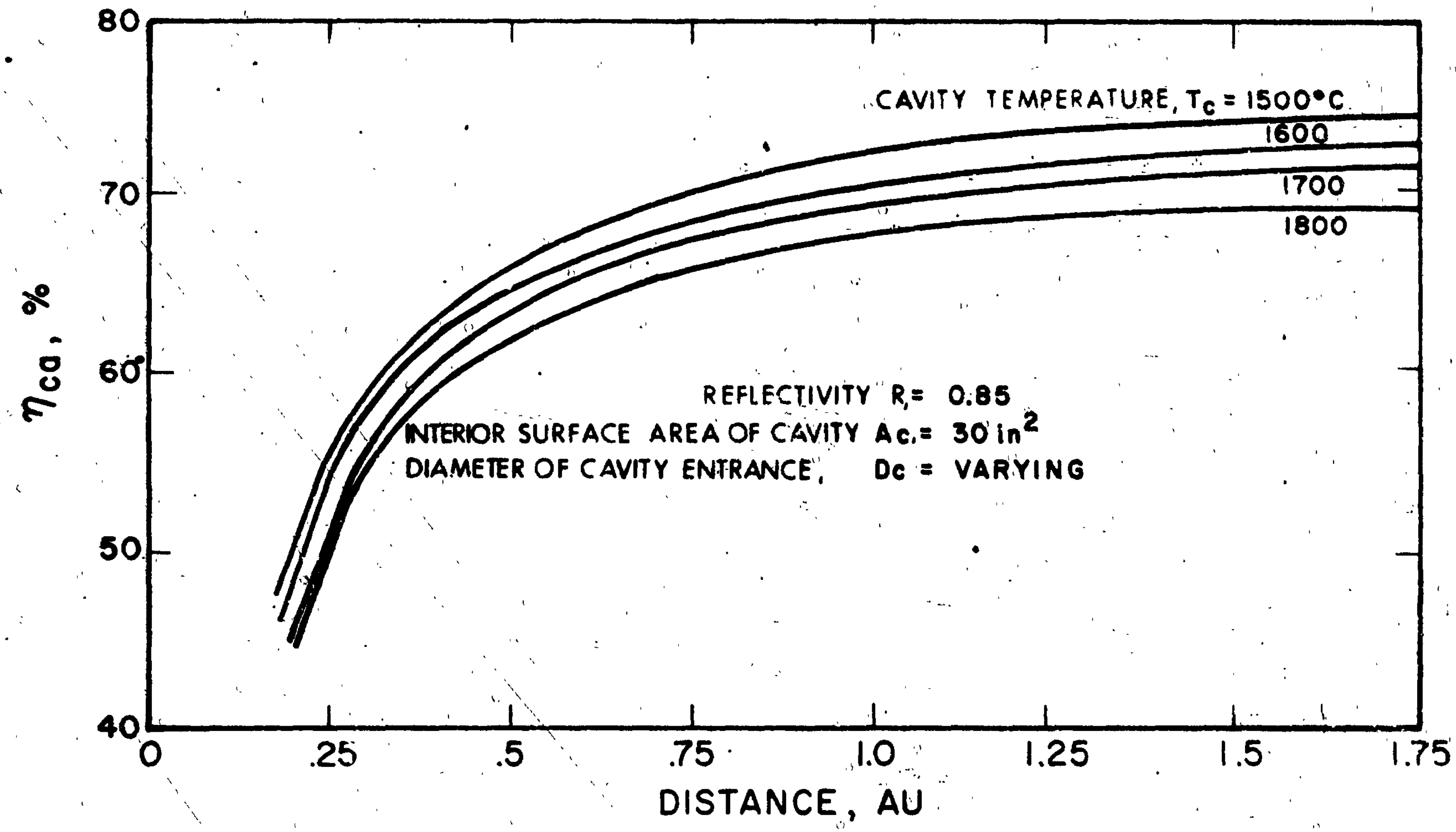


FIG. 4-4 OPTIMUM COLLECTOR-ABSORBER EFFICIENCY FOR 10-FOOT MIRROR VS DISTANCE FROM SUN

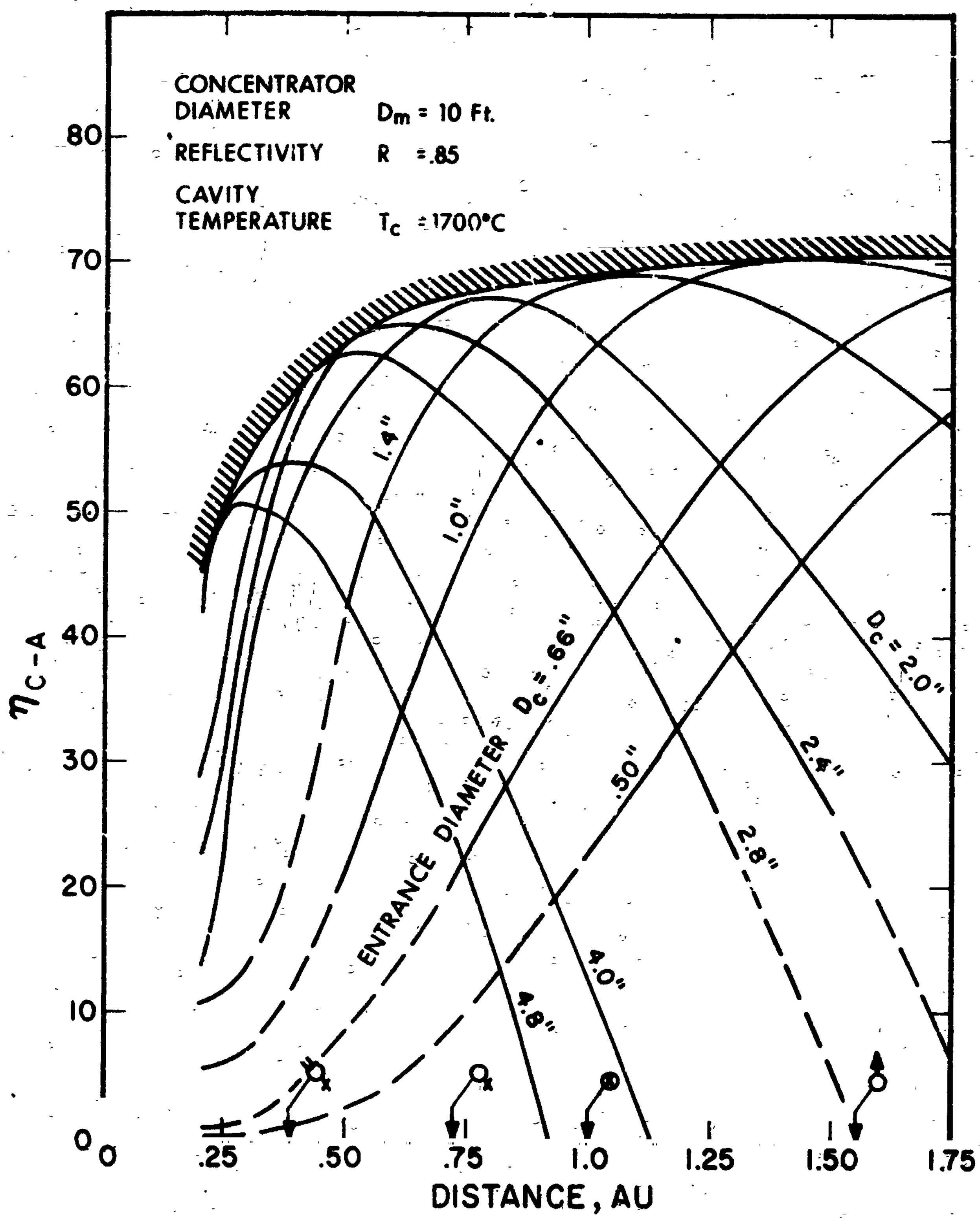


FIG. 4-5 COLLECTOR-ABSORBER EFFICIENCY VS DISTANCE FROM SUN AND CAVITY ENTRANCE DIAMETER

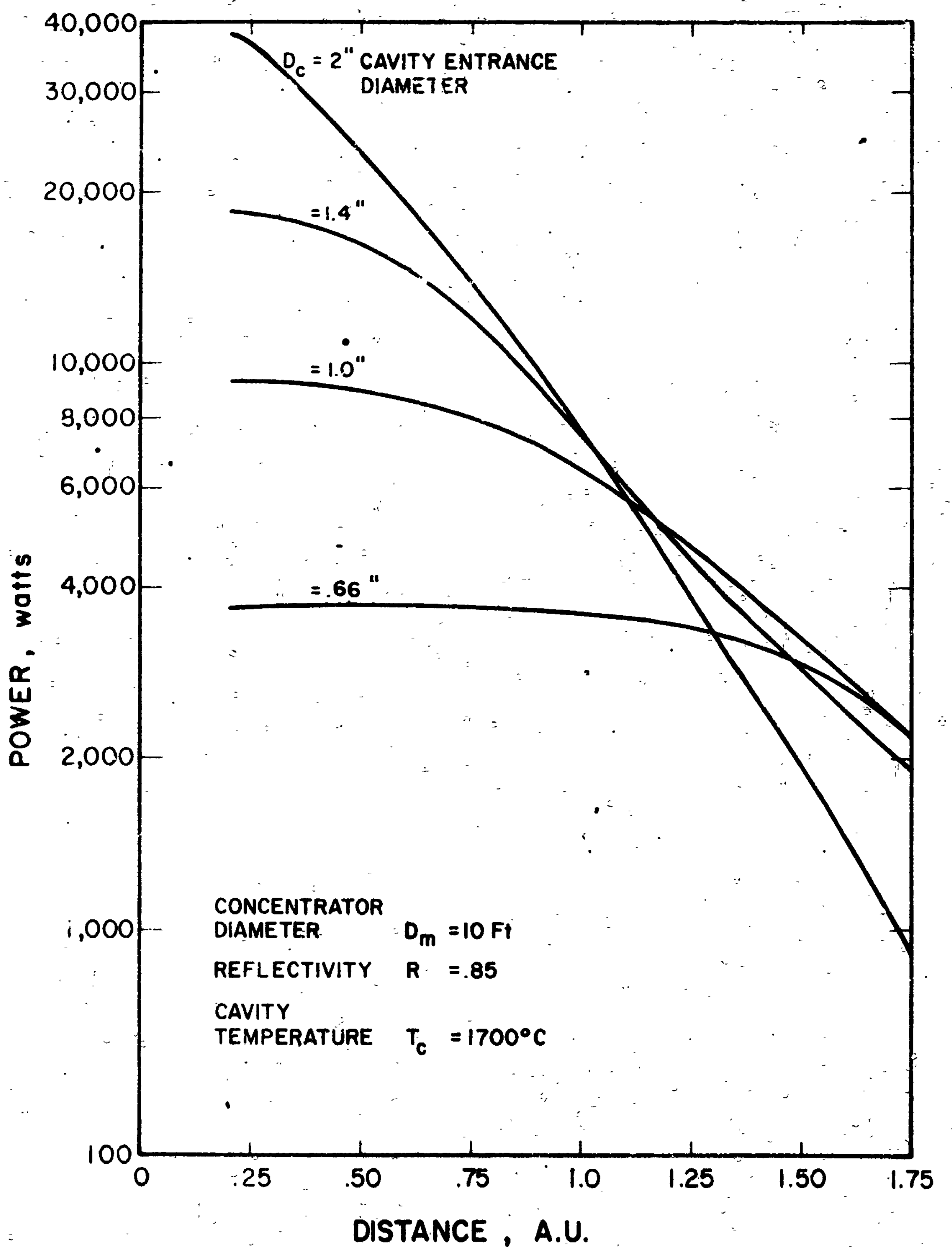


FIG. 4-6 THERMAL POWER INTO CAVITY VS DISTANCE FROM SUN

remainder of the analysis. Small variations for cavity temperature are apparent but not drastic.

Figure 4-4 shows the effect of distance from the sun on optimum efficiency. Close to the sun, the optimum collector absorber efficiency decreases drastically due to the increased size of the sun's image.

Figure 4-5 illustrates the variation in collector absorber efficiency using a constant entrance diameter (as in the solar probe case) with variation in distance from the sun. Figure 4-6 complements Fig. 4-5 and illustrates the thermal power input to the cavity as a function of distance from the sun. Figure 4-6 is extremely interesting in terms of the possibility of using passive flux control for a solar probe mission. Thus, for an entrance diameter of 0.66 inch, the power available to the thermionic diodes is essentially constant from earth to 0.2 AU with only small variations. A major problem would be the dissipation of the excess heat from the entrance cone of the generator.

Detailed reliability calculations were made for both the thermionic and photovoltaic systems. Table 4-III is a typical example of the reliability estimation for the power conditioning for each system. Minuteman type data was used for estimation of component reliability and, combined with estimates of the number of parts of each type and each module, an overall reliability was calculated for each case.

Figure 4-7 illustrates the specific weight of a solar-thermionic source and is typical of the weight calculations derived for the several missions and years. As shown, specific weight for the source decreases as a function of mirror diameter due primarily to the decrease in concentrator specific weight. Performance at 1700°C, due to higher efficiency, results in a smaller system weight than at 1400°C.

Figure 4-8 is a general curve illustrating the power output from a single mirror/generator system, under earth conditions, assuming 1975 component efficiency, as a function of mirror diameter and cavity temperature. For example, a 10-ft-diameter mirror can provide 1350 watts to the power conditioning.

TABLE 4-III

EXAMPLE OF RELIABILITY ESTIMATE FOR POWER CONDITIONING*

Photovoltaic System Component	n	$n\lambda(10^{-6})$
PS&L	1	0.199
Battery charger	1	1.316
Booster	1	0.656
Regulator	1	1.140
Inverter	1	0.397
Synch.	1	0.717

$$\Sigma_1 n\lambda = 3.925$$

$$R = \exp[-\Sigma_1' n\lambda t) = 0.966$$

Solar-Thermionic System	n	$n\lambda(10^{-6})$
DC/DC conv.	4	3.352
Shunt load	1	3.308
Battery charger	1	1.149
Battery conv.	1	1.353
Regulator	1	1.140
Inverter	1	0.397
Synch	1	0.717

$$\Sigma n\lambda = 11.446$$

$$R = \exp[-\Sigma n\lambda t) = 0.912$$

* 1) 1969

2) 500 km Earth Orbit

3) 1000 watt system

4) Power Conditioning is nonredundant

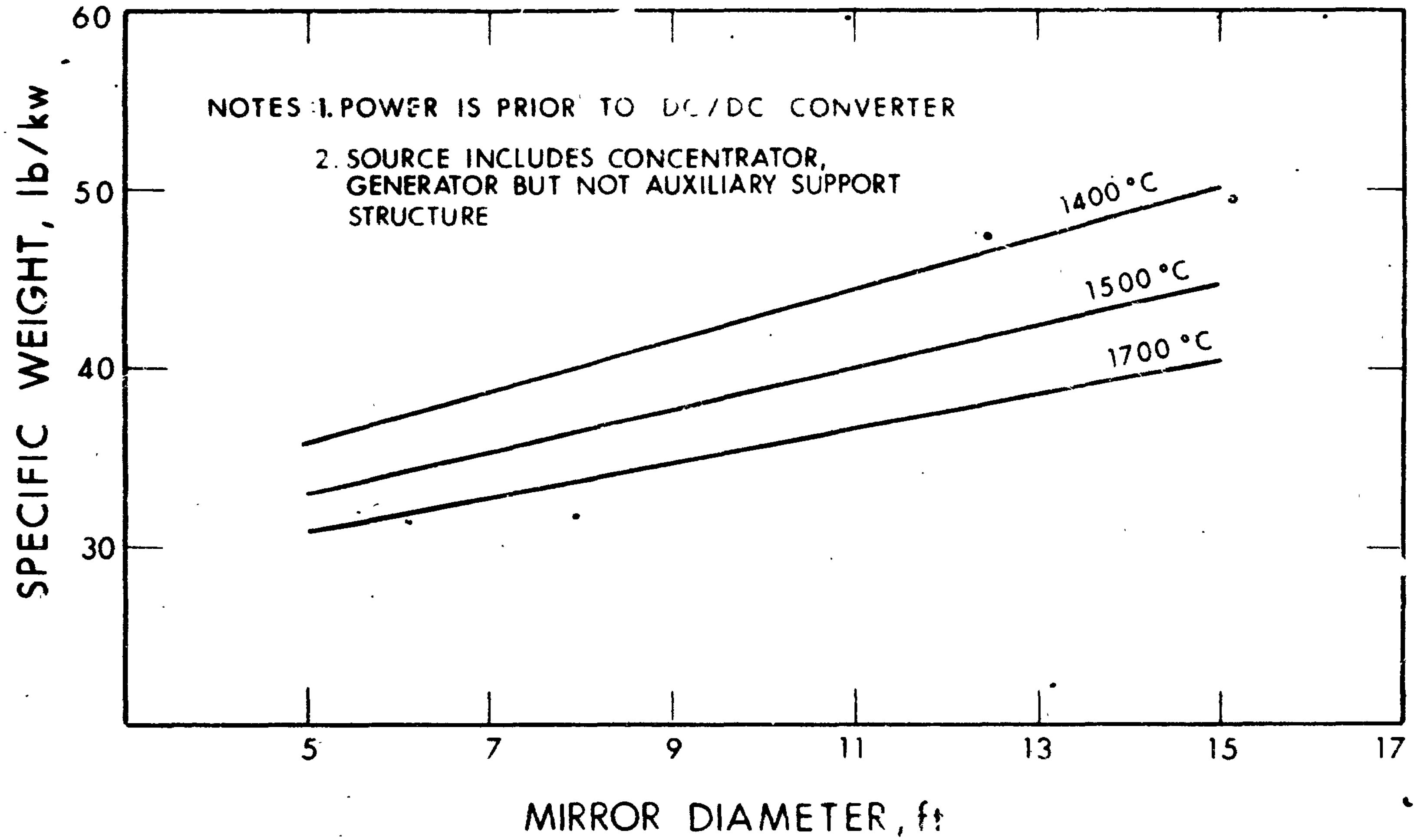


FIG. 4-7 SPECIFIC WEIGHT OF THERMIONIC SOURCE - EARTH, 1975

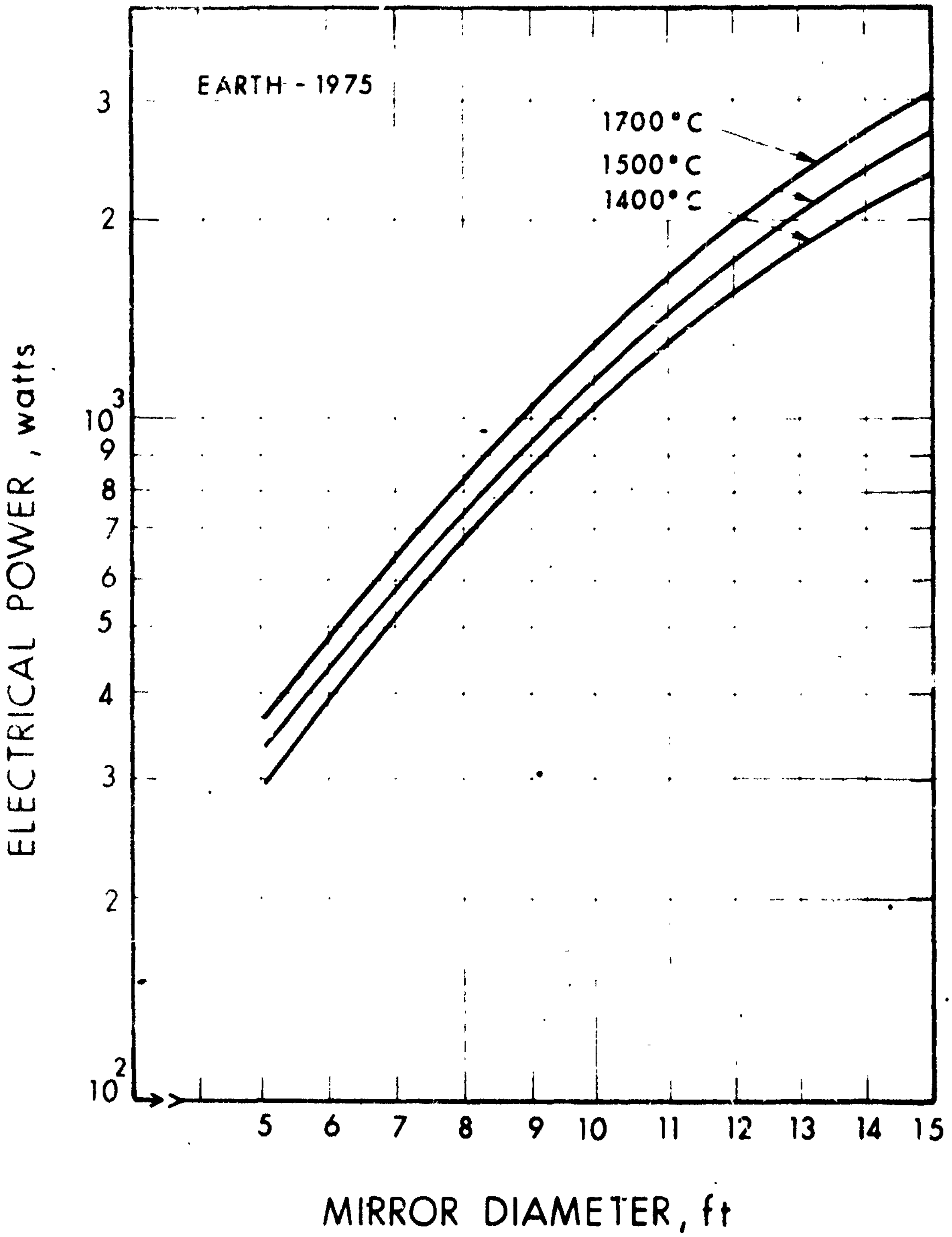


FIG. 4-8 THERMIONIC GENERATOR POWER OUTPUT FROM A SINGLE MIRROR SYSTEM

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5. PHOTOVOLTAIC SYSTEM DESCRIPTION

A block diagram of the photovoltaic system with redundancy is shown in Fig. 5-1. This system is quite similar to the Mariner IV power conditioning system except a series switching regulator is utilized rather than a booster regulator. The series switching regulator was selected since it can operate with greater efficiency over a wider input voltage range. The greater input voltage range is desirable, since it eliminates the need for solar panel voltage limiting that would require either zener diodes mounted on the solar panels or a similar radiator or some form of an active shunt regulator. In the redundant photovoltaic system, three batteries are employed such that any two can carry the entire system load. Redundancy is added to the solar panel by means of increasing its output capability by approximately 10 percent.

A critical item in the prediction of solar array performance is the assumption regarding solar cell performance. These assumptions are itemized in Table 5-I. As shown, from 1965 to 1977 the assumed cell efficiency will increase from 10.1 percent to 11.3 percent and useful thickness will decrease to 8 mils.

A second critical item in calculation of photovoltaic system weight is the assumptions regarding the solar panel structure. Detail calculations were made over a range of solar array surface areas resulting in estimated weights for systems employed in the ATLAS/CENTAUR and SATURN/CENTAUR. Results are summarized in Fig. 5-2 for 1975. Shown is the array specific weight (lbs/sq ft including mechanisms, cells, etc.) assuming the use of truss beam type structures similar to Mariner IV using thin gauge beryllium technology. The use of stronger materials and improved techniques results in relatively lighter weight for 1975. It is assumed that the

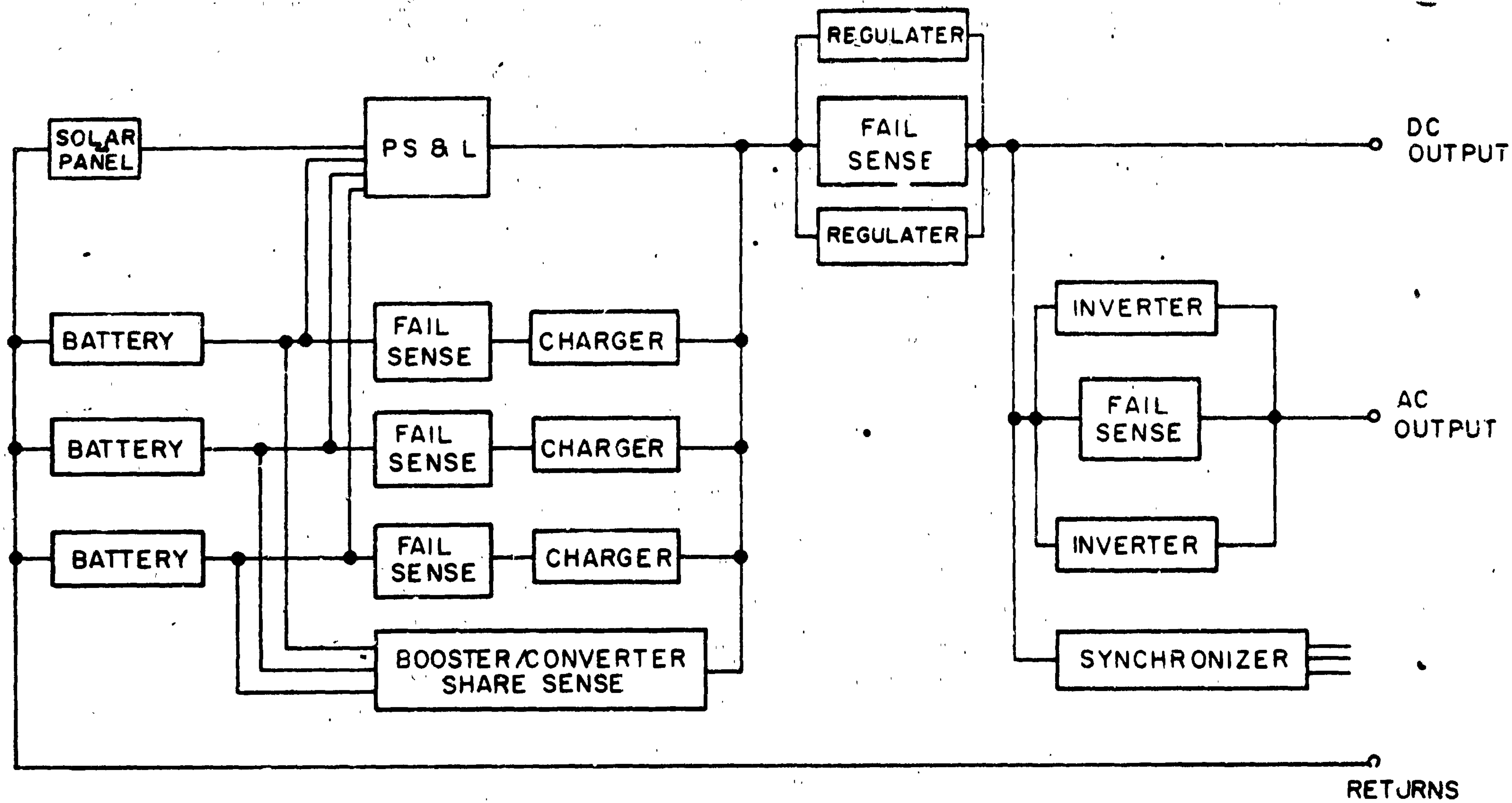

FIG. 5-1 PHOTOVOLTAIC SYSTEM BLOCK DIAGRAM WITH REDUNDANCY

TABLE 5-I

TABLE OF PREDICTED SILICON SOLAR CELL CHARACTERISTICS

	Present (1965)	1969-72	1972-77
Size	1 cm x 2 cm; 2 cm x 2 cm 3 cm x 3 cm	Same, also 1 cm x 13 cm, 2 cm x 13 cm	All before
Thickness	12 - 15 mil	12 mil	8 mil
Weight	0.09 gm/cm^2	0.08 gm/cm^2	0.045 gm/cm^2
Base Resistivity	1 - 10 ohm.cm.	1 - 25 ohm.cm.	1 - 25 ohm.cm.
Ohmic Contact	Solderless Silver-Titanium	Wrap Around Silver Titanium Solderless Cerium Titanium	Same
Power Output (55°C)	13 mw/cm^2	14 mw/cm^2	14.5 mw/cm^2
Total Cell Eff. (28°C)	10.1%	10.9%	11.3%

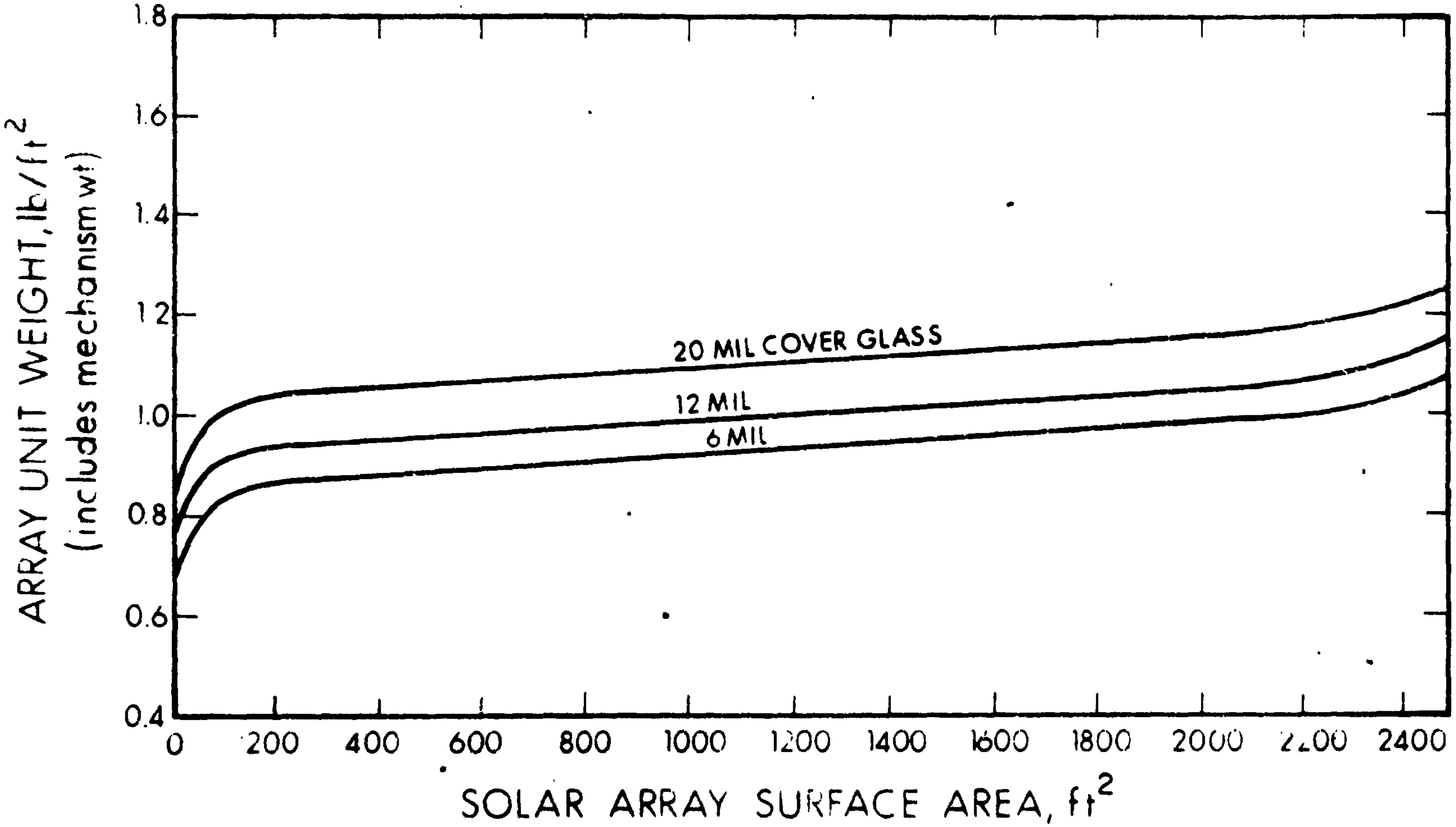

FIG. 5-2 SOLAR ARRAY SPECIFIC WEIGHT - 1975

solar panels are rigid and attached in a cantilever mode to the spacecraft at their base and an outboard location using a strut. A minimum resonant frequency of 2 cps and a retromaneuver loading factor of 3 g are critical assumptions which affect structural weight. The use of these assumptions results in a conservative estimate of photovoltaic structure weight. The effects of changing the assumptions and details regarding weight calculations are given in the Appendices.

Dark time or maneuver loads necessitate the need for additional power from the source charged batteries. Also, efficiency losses in the power conditioning require more surface area of the array (either thermionic or photovoltaic). The ratio of solar panel raw power output to conditioned power is illustrated in Fig. 5-3 (known commonly as an "F" factor). The 500 km earth circular orbit requires an F factor typically of 2.75. The smallest F factor is for the solar probe lunar station and 50 hour orbiters where a value of 1.5 is required.

Considerable effort was expended on estimation of solar panel weight performance as a function of mission time, distance from the sun, cell efficiency, cell series resistance, and other factors. A large number of curves illustrating this effort are contained in the report text.

Figure 5-4 illustrates equilibrium solar panel temperature vs distance from the sun for varying angle of incidence to the sun. As shown, solar panel temperature will increase above a 250°C value at 0.36 AU with the panel normally oriented towards the sun assuming the absorptivities and emissivities shown.

Figure 5-5 illustrates the typical change in EI curves obtained on operating panels. Of particular interest is the effect of assuming 0.4 ohm series resistance on the solar cells as opposed to the effect of assuming zero series resistance. As shown in Fig. 5-5, using typical equilibrium temperatures, the decrease in power output of the

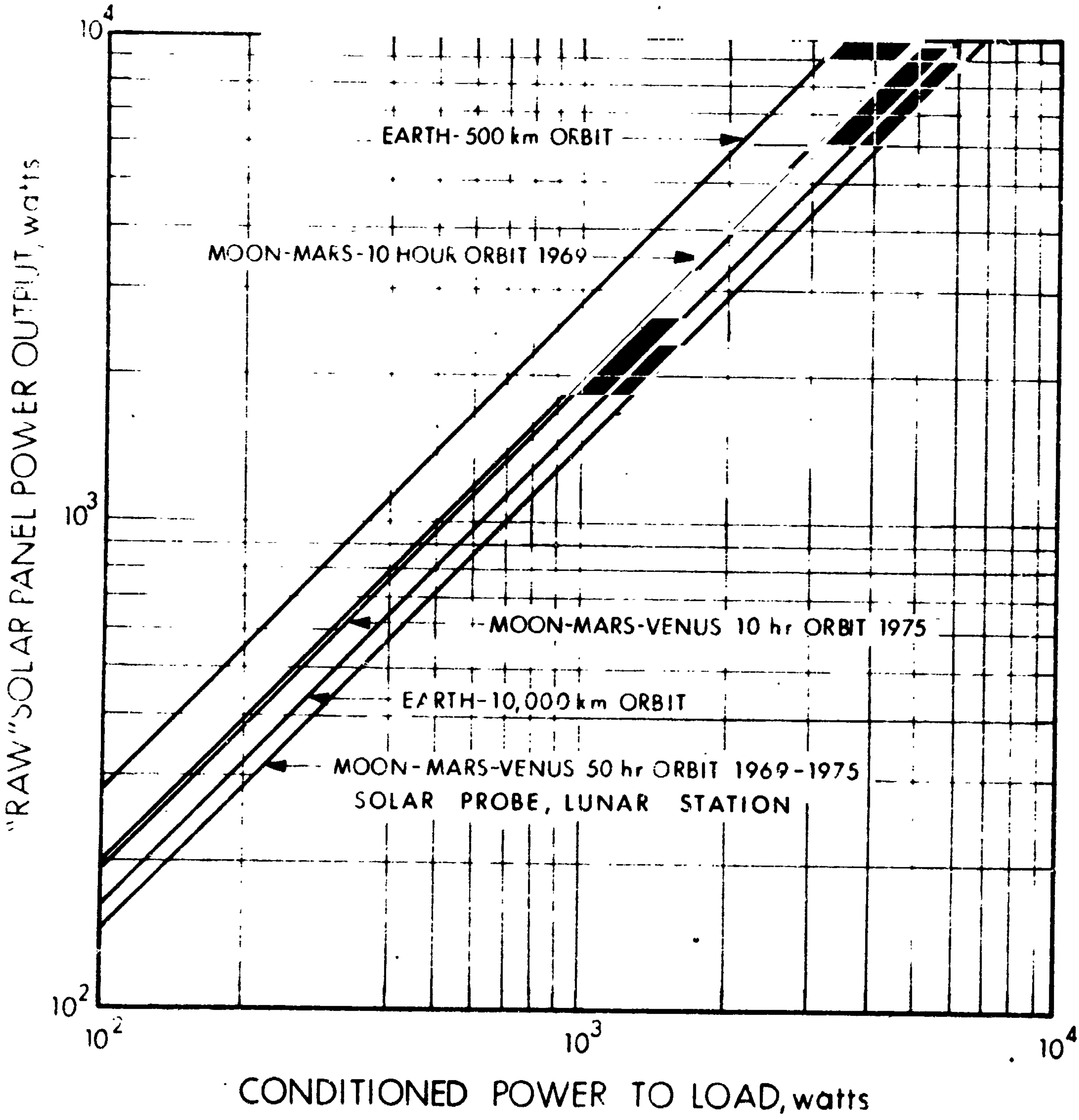

FIG. 5-3 CONDITIONED POWER FROM SYSTEM VS REQUIRED SOLAR PANEL RAW POWER OUTPUT

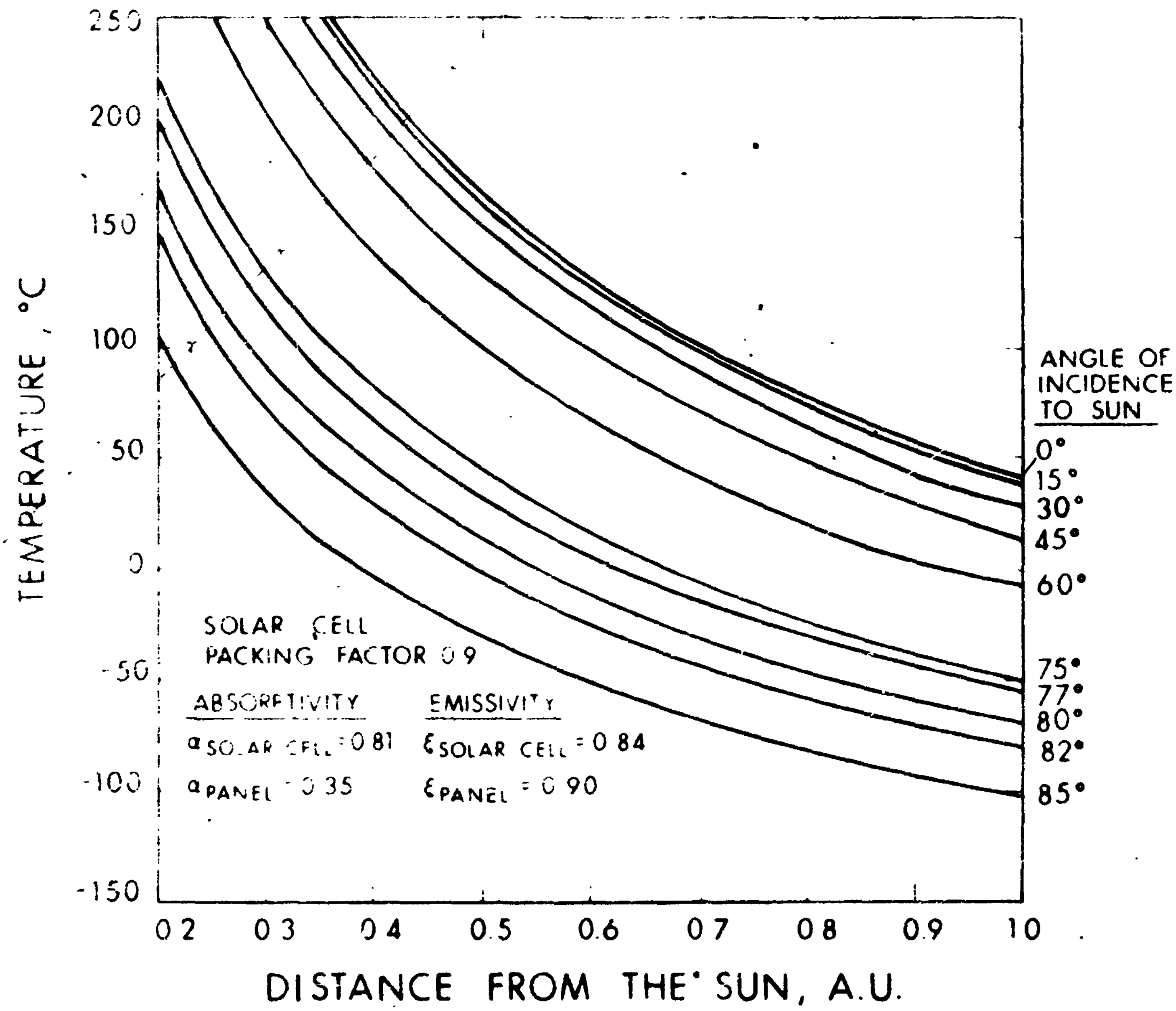

FIG. 5-4 SOLAR PANEL TEMPERATURE VS DISTANCE FROM THE SUN FOR VARYING ANGLES OF INCIDENCE

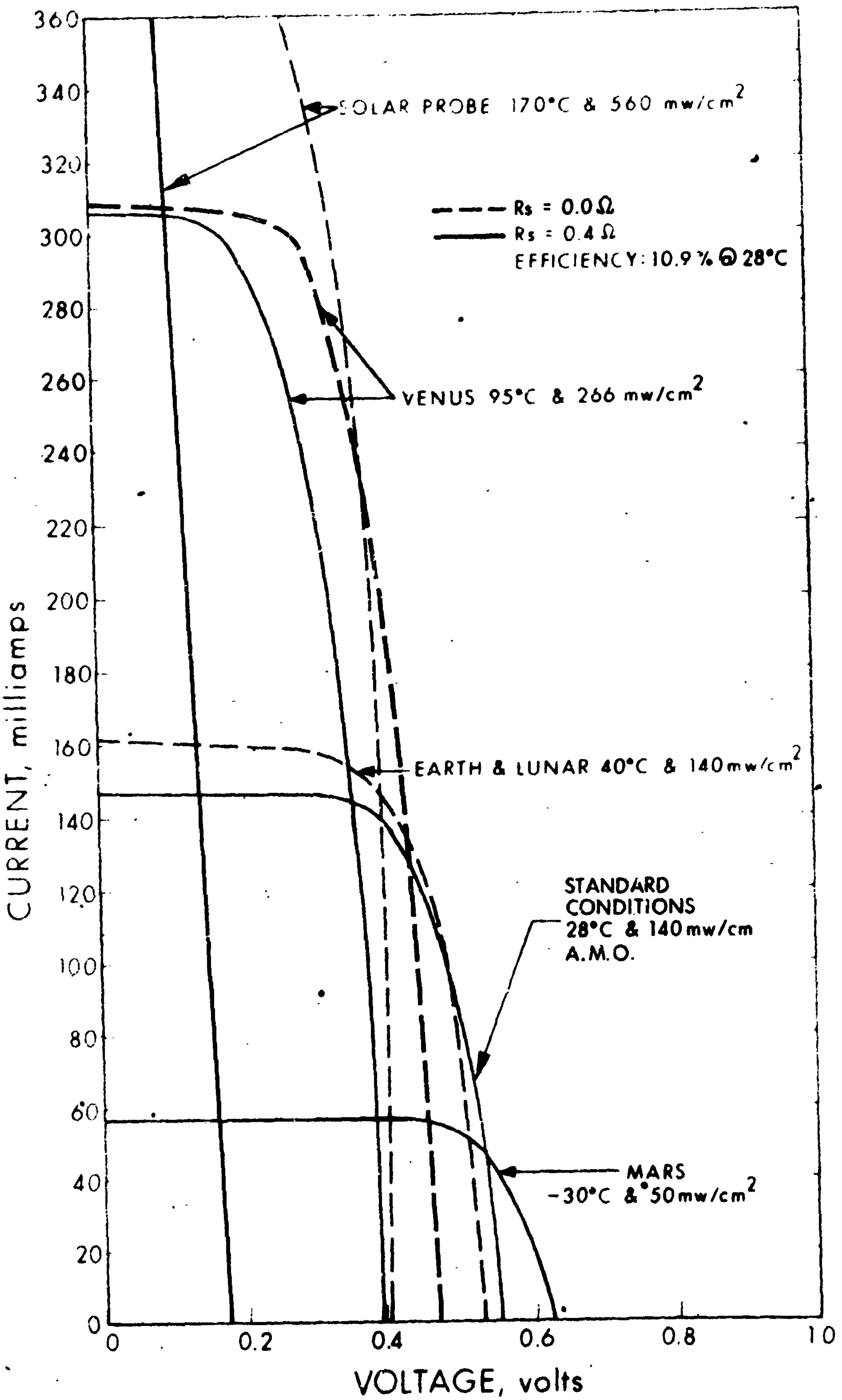

FIG. 5-5 TYPICAL VOLTAGE-CURRENT CHARACTERISTIC CURVES FOR VARIOUS MISSIONS

panel and the change in shape of the EI curve is drastic depending on what is assumed for series resistance. Calculations used for deriving system weight were based on the changing EI curves typically illustrated in Fig. 5-5.

Figure 5-5 illustrates the need to carefully investigate the design of the solar cells used for solar probe missions. At the present time, no solar cell is available in quantity which has series resistance lower than about 0.4 ohms. It is possible to create a solar cell with nearly zero ohms resistance at decreased efficiency. Development of such a solar cell wou d greatly benefit the use of a photovoltaic system for a probe mission.

Figure 5-6 illustrates some of the radiation degradation calculations performed for the study. Percent power remaining at the end of a one-year mission is given as a function of coverglass thickness for several missions in 1969 and 1975. An inverse square relationship was assumed for the solar flare degradation. Degradation due to Van Allen radiation at Earth was based upon current aerospace models for particle flux spectrum and density. Figure 5-6 illustrates the difference in radiation degradation which occurs using a maximum solar flare intensity year (1969) and a minimum year (1975).

Figure 5-7 shows a typical mission time - power history using selected coverglass thicknesses, assuming zero and 0.4 ohm series resistant cells. As shown, for the Venus mission, the power per unit area can vary drastically depending on cell assumptions. The power output changes due to temperature equilibriums, radiation degradation and solar intensity change.

Figure 5-8 shows a second typical mission time power history for the Mars case. Figure 5-9 illustrates the raw power per sq ft vs angle of incidence for varying distances from the sun using a 20-mil coverglass and two different values of series resistance of the cell. The effect of series resistance on solar array performance is dramatic.

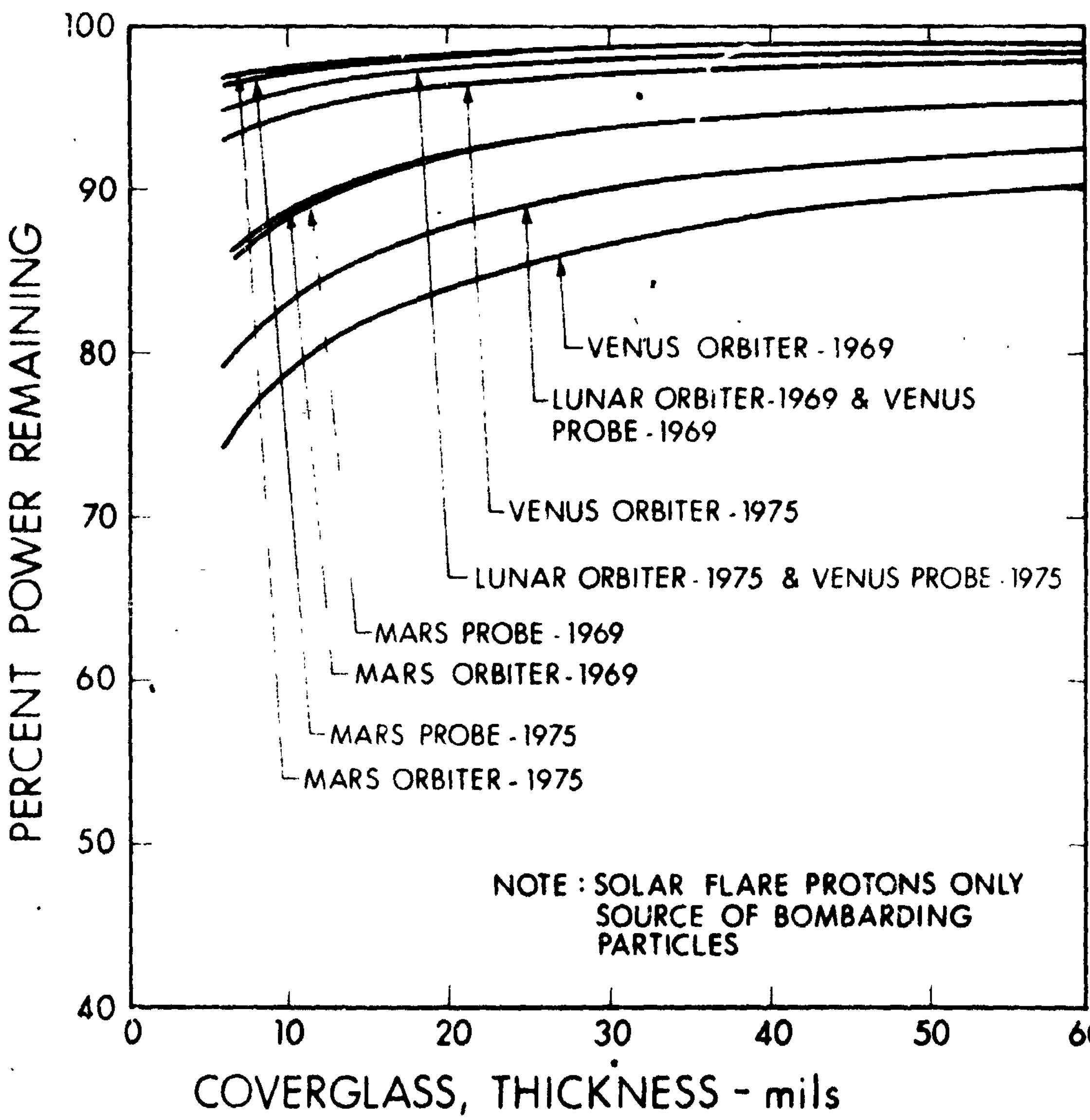

FIG. 5-6 PERCENT POWER REMAINING AFTER RADIATION BOMBARDMENT AT END OF MISSION VS COVERGLASS THICKNESS

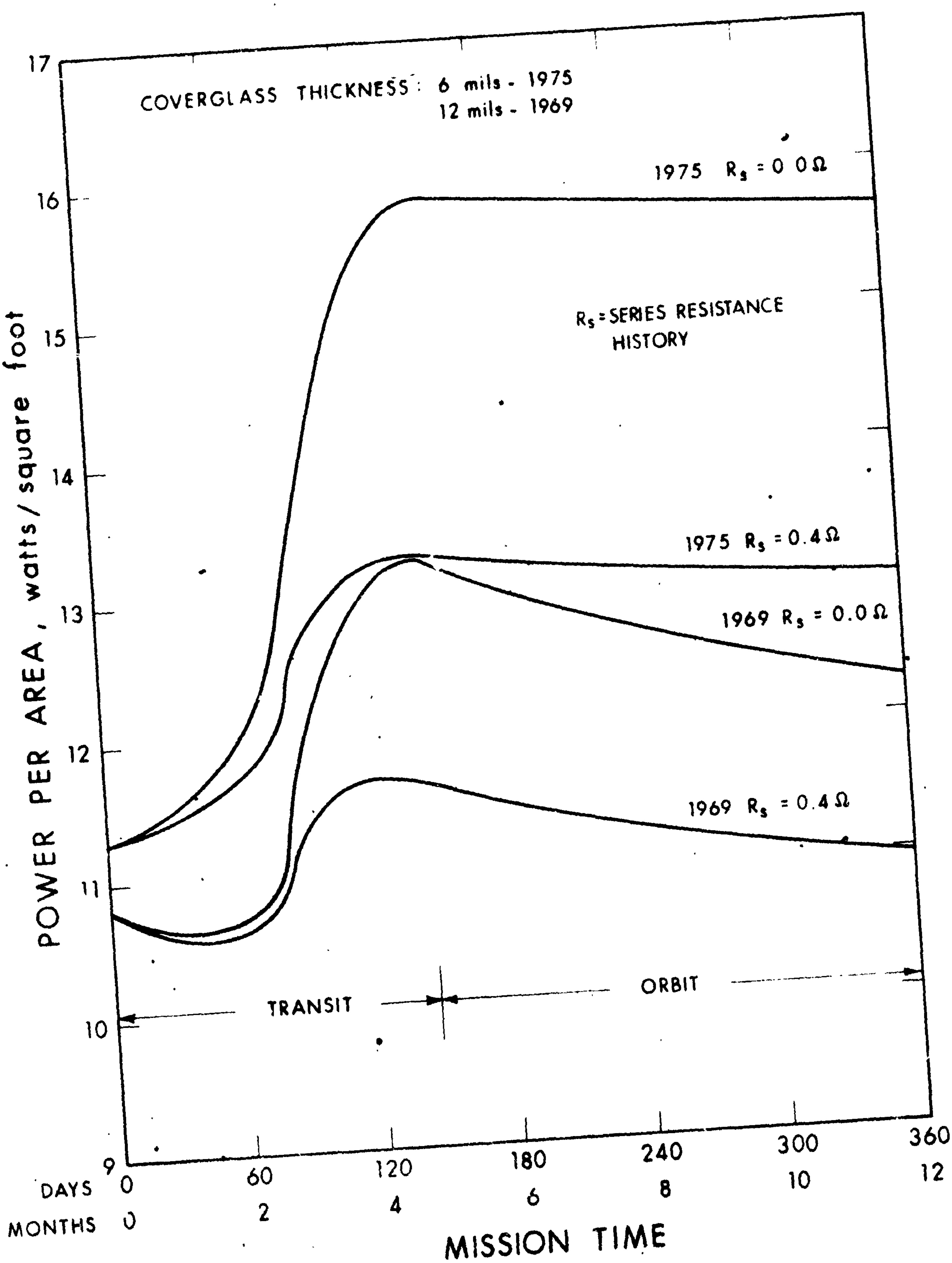

FIG. 5-7 VENUS MISSION PHOTOVOLTAIC ARRAY POWER OUTPUT WITH TIME

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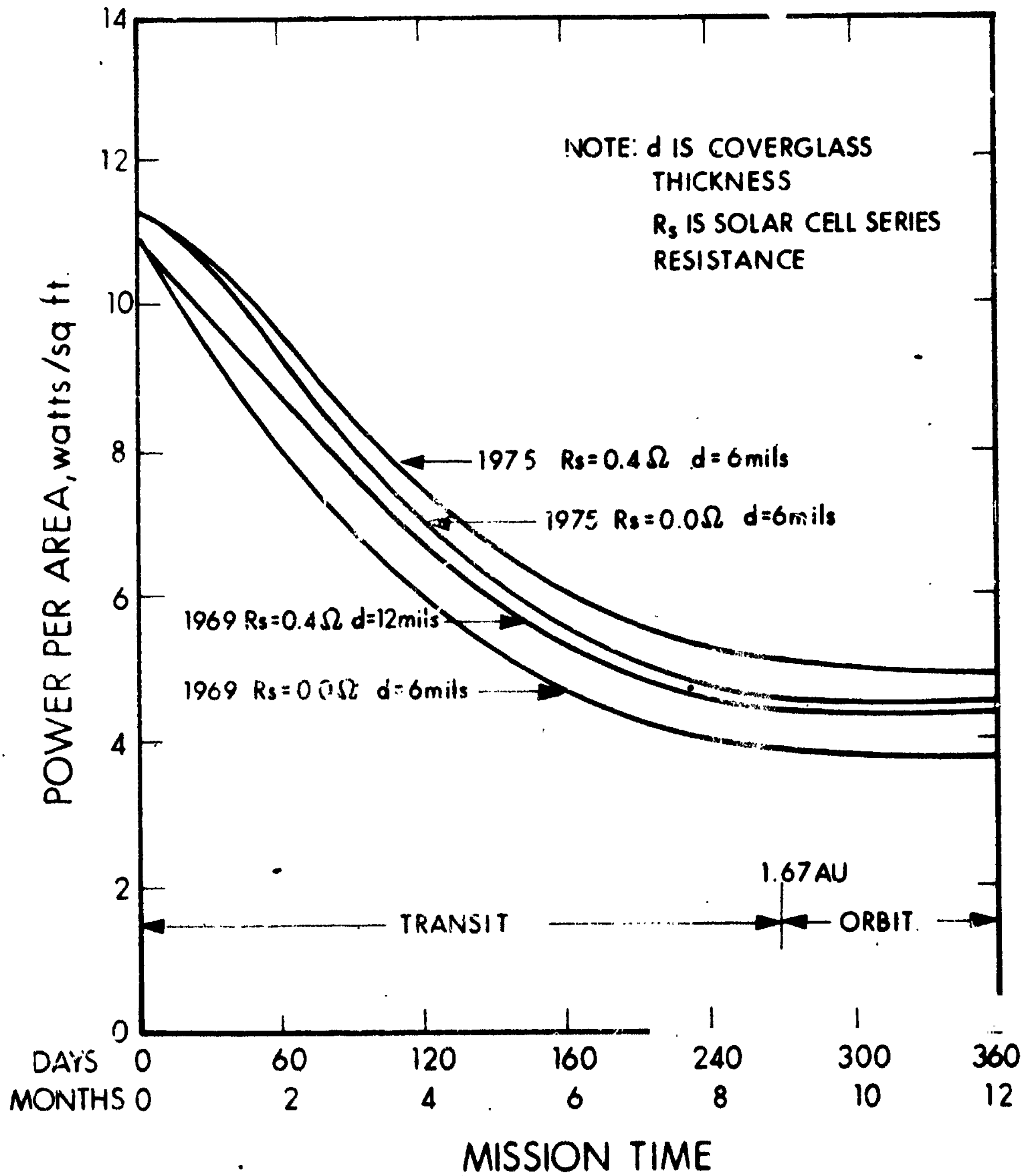

FIG. 5-8 MARS MISSION PHOTOVOLTAIC ARRAY POWER OUTPUT WITH TIME

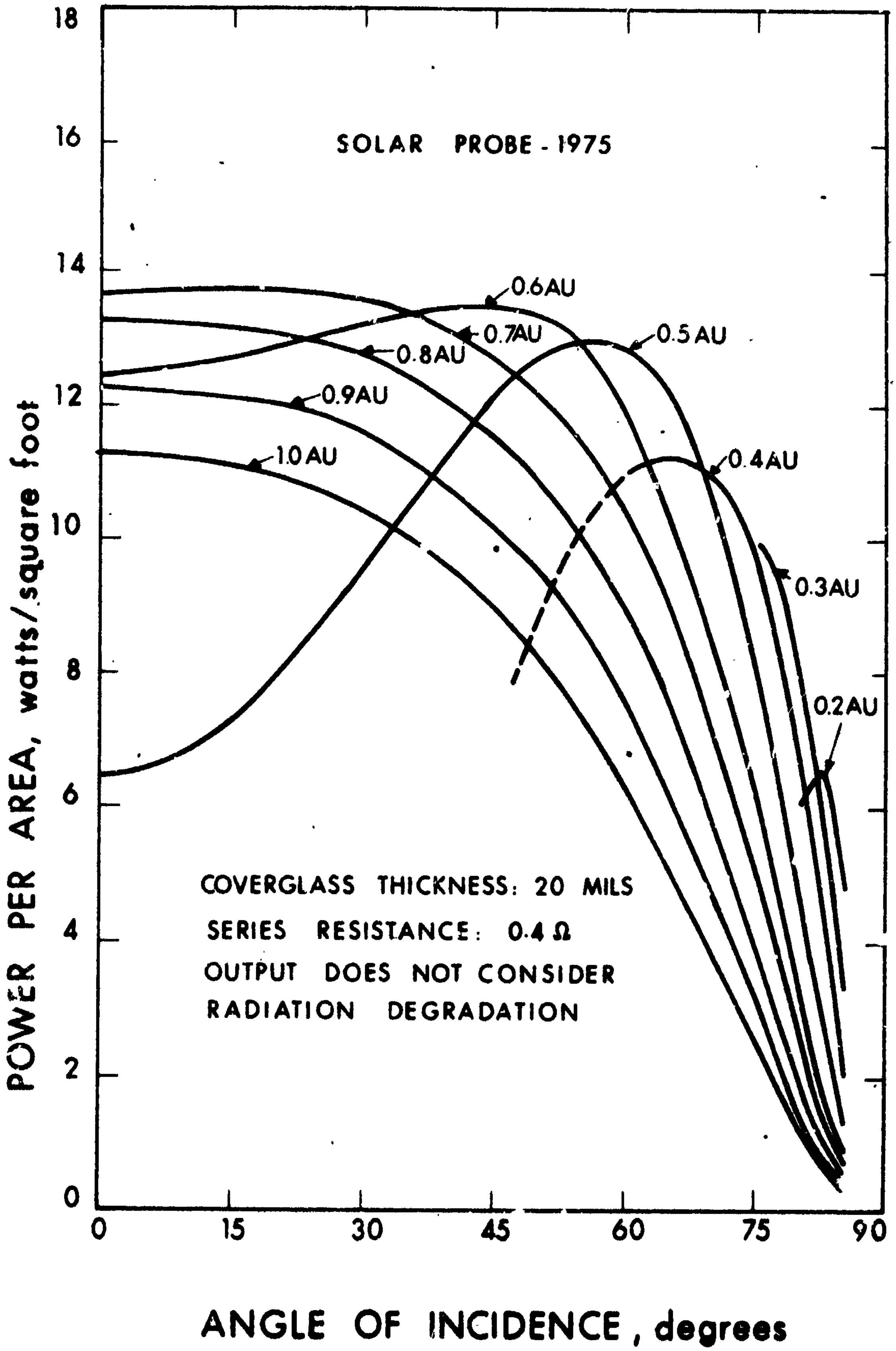

FIG. 5-9 PHOTOVOLTAIC ARRAY RAW POWER PER SQUARE FOOT VS ANGLE OF INCIDENCE FOR VARYING DISTANCES FROM THE SUN

Thus, with solar cells presently available, the performance at 0.2 AU is clearly limited and the solar array must be inclined toward the sun an angle of approximately 82 degrees. With future cells of zero series resistance, power output is much greater at 0.2 AU and optimum angle of incidence decreases.